NASA TMX-50357

N-107,222

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60P

N63 84915

Code 5

THE NIMBUS SATELLITE AND ITS COMMUNICATION SYSTEM

SEPTEMBER, 1961

Invited lecture presented at ~~the~~ Calif. U., Los Angeles
University of California,
Los Angeles, California

NASA — GODDARD SPACE FLIGHT CENTER —
GREENBELT, MARYLAND

(NASA TM X-50357)

THE NIMBUS SATELLITE AND ITS COMMUNICATION SYSTEM

by

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INTRODUCTION

The Latin word Nimbus means cloud, raincloud, halo, and fame. It is also used as a meteorological term for any rain-carrying cloud adopting one of the specific meanings of the ancient word. The same name was given to the forthcoming Meteorological Satellite series, a rather complex spacecraft which will gather more meteorological data in one orbit than the present meteorological network compiles in days. As a source of inputs to this network, it will deliver its information together with a multiplicity of others and will thus close many information gaps in existence today. Beyond these objectives the spacecraft is a platform for the conduction of a series of experiments, primarily those which require a stabilized platform in a polar orbit. It is hoped to encompass the full range of meanings of the classic word using the spacecraft as a tool to produce results and meet the objectives set.

Proof of the usefulness of television information was delivered by earlier satellites making it mandatory to choose many characteristics of the satellite so that television and other sensory information can be optimized. This readily leads to the choice of a polar circular orbit since the rotational movement of the earth provides a means for complete geographical coverage by the utilization of the television system for taking pictures of cloud cover. System considerations on the spacecraft power supply naturally show considerable simplifications when attitude of the solar power supply with respect to the sun is maintained as constant. This feature can be achieved when the orbital plane always contains the earth-sun line.*

The earth has the shape of a nonspherical body, or geoid, the equatorial diameter of which is 43 km larger than the polar diameter. Due to this, a nonspherical gravity potential field is generated and the angular momentum vector (normal to the orbital plane) precesses around the earth axis. The orbital inclination remains constant but its precession rate depends on its value. This precession is normally referred to as the rate of nodal regression of satellites.

Satellites launched into circular orbits towards the southwest quadrant (retrograde orbits) experience a regression of the modes from west to east. By choosing an inclination 80.11° from the equator, the rate of regression of the modes will be 0.98° per day, exactly that of the mean sun relative movement around the earth in terms of celestial

*Bandeen, W. R., "Considerations in the Determination of an Ideal Orbit for Nimbus," NASA Technical Note, D-1045.

coordinates. With the selection of this launch mode, the features of a near-polar orbit is maintained, and superior television coverage of clouds and of terrestrial radiation will be obtained. Other considerations necessary for obtaining optimum data from the television system indicate the desirability of an orbital altitude of approximately 1000 km. The rate of regression depends not only on inclination but also on altitude. An error analysis for the system based on launch vehicle performance margin shows that errors of 1^{o} in inclination will cause an 18^{o} deviation from the mean sun position after a half-year of orbital life with altitude errors of 70 km corresponding to 6^{o} deviation for the same period of time. Orbital characteristics for the Nimbus satellite may thus be established from calculations and celestial data already accumulated.

Basic Spacecraft

Weather observation from a satellite demands the sensing of atmospheric phenomena. This is best performed by a stabilized platform viewing the earth continuously from one specific area, in this case the base plate; consequently, the satellite must be controlled in all three axes. Error signals are generated by two horizon sensors* and a sun sensor which are used as computer inputs to control gas jets and inertia wheels. During the satellite eclipse a rate gyro reference replaces the sun-sensor error signal. Resetting of the gyro occurs when Nimbus crosses from the earth's umbra into sunlight. Gas jets are mounted in such a fashion that control in three axes is possible to $\pm 3^{o}$ from the local vertical direction. Sufficient gas supply is carried to sustain operation for 1/2-year. The inertia wheels in conjunction with the gas systems will stabilize to $\pm 1^{o}$. By design the flywheels will primarily function to control periodic disturbances, and when maximum speed is reached the gas systems will unbias the inertia wheels. Angular rates are controlled to $\pm .05$ degrees per second in all three axes, a value which enters directly as a design parameter into all optical scanning systems. The feature of a retrograde orbit results in considerable simplification in the control system. The present state of the art dictates that solar energy and photovoltaic energy converters be used. These solar energy units (solar cells) will have constant power input when the sun angle is maintained at 90^{o} to their surface and as long as illumination exists. The orbit selected due to the sun rays will always be nearly parallel to or in the orbital plane over the projected life time of 1/2-year so that only the rotational motion of the satellite - once per orbit - generated by the control system for the satellite must be counteracted by the control system for the solar paddles.

*Control system designed and produced by MSVD, General Electric Companies, under contract to NASA/GSFC.

A brief discussion of the satellite configuration is appropriate at this time so it can be referenced in the following discussions (Figure 1). Stabilization is enhanced by the earth gravity potential field when a body is designed to the shape of a dumb-bell such as our moon. This effect aids the control system but it is not taken into its design thus leaving a small additional margin of safety. Further conceptual design considerations of the spacecraft are related to the degree of flexibility. Spacecrafts serving as research tools must provide a high degree of flexibility permitting change of subsystems readily. This guiding principle will become evident throughout the whole spacecraft, particularly in controls, power supply, routine telemetry, and antenna system. In agreement with this reasoning, the control system and sensory system are physically separated as shown in Figure 1.

The control system with its error detectors is contained in an octagonally shaped container from which the solar paddle drive shaft protrudes. The top carries the interrogation antenna. Except for wire connections including primary power the system is independent from the spacecraft including its thermal design. Slip rings are provided on the paddle shaft for feeding power to the batteries, the sun sensor voltage and for signals to the paddle. The resulting mechanical and electrical interface between paddles and the control system is extremely simple. The hexagonally shaped control system container measures 22 inches from corner to corner and is 18 inches high, and total weight is 104 pounds. Truss members 48 inches long connect it to a toroidal section carrying sensors and its associated equipment. This sensory ring is divided into 18 compartments each being 6 by 8 by 18 inches. All equipment is designed to fit into these standard volumes or into subdivisions by quarters of the 8 inches in dimension or by dividing 18 inches by two. Eight increments provides sufficient flexibility for modular design of electronic and some mechanical components. Optical equipment, such as scanners, cameras, and passive detectors or other bulky equipment including tape recorders, find space in the cylindrical section within the torus or underneath the torus. The entire base area can so be used for instrumentation. The engineering advantages of this design being: (1) each instrumentation box can easily be replaced, (2) subsystems can be changed from flight mission to flight mission, (3) the control system can be developed almost independently of the spacecraft, (4) the power supply can be substituted against a higher power one or nuclear one, and (5) finally the choice of experiments is almost unlimited. Torus and truss weight is approximately 60 pounds using magnesium where possible. The independence of controls and sensory torus is further enhanced by making the thermal properties of those two sections as independent as possible, the thermal power flow through the truss is less than 2 watts. Both sections contain active temperature controller devices designed to operate like venetian blinds. They serve as energy valves permitting heat to radiate towards the cold sky when they are open and conserve heat when they are closed. An error detector is located on each side of the 18 boxes mechanically actuating the blinds so that an opening is assured corresponding to a 25°C. nominal temperature.

Testing of the spacecraft and calibration of the sensors will not be discussed in this presentation. However, many design features although have been influenced by the testing requirement. Flight spacecraft will be tested at 7 g's r.m.s. white noise (gaussian) extending from 20 cps to 2000 cps. Approximately 30 days will be spent by the spacecraft in a thermal vacuum test, half of the time at 45°C and half at +5°C. This represents a margin of 10°C beyond the maximum error of the thermostat.

Calibration is performed by mounting appropriate calibration equipment with test patterns or other targets in an adapter section connecting the spacecraft with the rocket carrier. This procedure, more accurately termed a "check of calibration", since primary calibration is performed prior to installation of the system in the spacecraft, must be performed in vacuum as will be described later. It consumes approximately 30 days in a space simulator chamber possessing cryogenic and high vacuum capabilities.

Power Supply

The power supply is a solar conversion type that has been designed as a separate unit, but is closely associated with the design of both structure and controls. Solar power has been chosen for best weight economy and because it has sufficient reliability and longevity to be used for extended periods of time in a space environment.

When the Nimbus spacecraft is orbiting at 1000 km in a circular orbit, it will spend 69 minutes in sun light and 38 minutes in the earth umbra. Referring to Figure 1 it can be deduced that 41,040 cm^2 of silicon cell surface area is available to intercept solar energy corresponding to 5.7 K watts during sun illumination. The fraction that is finally available during the entire orbit is subject to a large number of parameters which must be taken into account. Efficiency of silicone cells can be selected to be better than 11 percent for a large production yield even when 2 by 2 cm cells* are produced. However, if a more detailed analysis is desired the efficiency degradation is due to increasing surface temperatures, various types of filters used to reject certain radiation heating effects, and space environment erosion.

The various subsystems require a multiplicity of voltages with many conflicting voltage and stability demands. The only practical solution to solving these problems is in the employment of DC to DC

*International Rectifier Corporation under contract to RCA converters with auxiliary regulation. Input voltage for these circuits is determined by the breakdown voltage of transistors for the maximum and a desire to operate as close to that maximum voltage as possible for optimum conversion efficiency. These reasons, and the experience gained from the development of a multiplicity of other spacecraft power supplied in the TIROS series, led to the choice of -24.5 volts and a regulation of ±2 percent.

The quantity of silicon cells needed to gather energy lends itself uniquely to redundant designs by series and parallel connection of the cells, and an arrangement of blocking diodes so that strings producing low voltage do not load those producing higher voltage. The cell efficiency and losses incurred due to interaction of cells and losses inherent in the diodes, and allowing further for the spectral response of the cells at orbital altitude, 440 watts can be obtained from the cells when the paddles are illuminated at their mean temperature of 30°C. When Nimbus just leaves the earth umbra, the available power will be lighter and continuous heating in sunlight will decrease the power output. To accomplish a favorable mean temperature, cells are coated with a filter material to provide the appropriate absorbtivity to emissivity ratio to receive light in the spectral region where the cells are sensitive and reject infrared radiation components of the spectrum. A glass cover provides mechanical protection against meteorite erosion and serves as a base for the filter coating, the total weight of each solar paddle is 23 pounds.

The output voltage from the silicon cells will not be constant thus charging of the storage batteries must be regulated by charge regulators (Figure 2). Sealed nickel-cadmium cells, that are practical for operation in a vacuum, are used to provide the energy storage capability for the spacecraft. Further advantages of these cells are good energy to weight ratio, convenient temperature operating range, and high cycling capability. Nevertheless, the batteries are an element of lower reliability than solar cells and redundancy must be provided. The power supply designer can choose the depth of discharge, extra capacity by adding more battery strings and separation of battery strings by appropriate circuitry. For Nimbus, 4.5 ampere hour cells* have been chosen and will be discharged 15 percent of their full ampere hour capacity. Eight separate and independent battery packs are provided where the total load can be satisfied by 75 percent of the capacity or 6 batteries. Charging the nickel-cadmium batteries not only depends on the state of battery discharge but also on the temperature and gaseous pressure within the cell. High-temperature and high-pressure destroy the cells and since additional charge generates a run-away situation in this direction, sensors at each battery string operate a switch and reduce the regulator output to a trickle charge rate sufficient to maintain the state of charge without further

*Sonotone Corporation - under contract to RCA temperature buildups or significant gas generation. The regulators themselves are conventional current-series regulators of the DC amplifier type, limiting the current initially to 1.5 amperes. The trickle charge rate is kept to 0.5 ampere.

When the batteries are in the trickle charge state due to excess temperature or pressure, no power could be supplied to the load through the discharge regulator. In order to overcome this condition, a diode is supplied to bypass the regulator and an overvoltage protection of the unregulated power bus guarantees that a safe limit for the discharge regulator is not exceeded. This arrangement also provides power to the load under sun illumination when all batteries have failed. Each of the eight units has a series discharge voltage regulator. A voltage reference and feedback amplifier common to all regulators senses the output voltage and provides a feedback signal to the discharge regulator. At the same time this regulator is driven by a forward regulator so that batteries in a higher state of charge share more of the load than batteries in a lower state of charge. A redundant reference unit and feedback amplifier guards against malfunction. Two voltage comparators sample the low and high voltage limits and compare them with the regulated bus voltage. If one of the limits is exceeded, a relay switches the redundant feedback reference amplifier into the loop. Separate fuses disconnect modules from the output when energy would be fed into them rather than from them.

This power supply can supply a load of over 200 watts for the entire orbit. In preview of the actual instrumentation and subsystems it shall be stated that the demand during earth-day is 206 watts; during earth-night 170 watts; and during interrogation (day) 400 watts, and interrogation (night) 342 watts.

Having presented the spacecraft configuration it appears convenient now to describe its ascent into orbit. A Thor-Agena B will lift the payload to a transfer ellipse. After a second Agena B burn a rather precise circular orbit will be achieved and the rocket carrier will be guided out of the satellite orbit. Following separation, the solar paddles (Figure 1) which are still folded against the truss are unlocked and driven open by a motor until they finally latch in the open position and assume their normal function. After appropriate time delays, the control system starts to orient the satellite to assume its tasks in space. In the framework of these constraints presented here we will proceed to look at the subsystems and their communication requirements.

It is obvious that meteorological information must be related to geography to be of any use. Orbital determination and accurate time are therefore very important. Required tracking accuracy is compatible with the Minitrack system*. This system uses a 136.5 mc beacon in the

*Mengel, J. T. - "Tracking the Earth Satellite, and Data Transmission by Radio Proceedings," IRE, 1956 - pp 755-760 satellite and an interferometer arrangement in a ground network. Modulation on the beacon does not interfere with the tracking network as long as no sidebands lower than 1000 cps are adjacent to the carrier and sufficient carrier power is provided under all modulation conditions.

At this time the reader will well appreciate the complexity of the spacecraft making a justification for analytical telemetry superfluous. Engineering evaluation of the system in orbit provides the basis for more reliable designs in the future and beyond it permits the user of the spacecraft to prolong its life by turning malfunctioning subsystems off through the command system or by replacing redundant modules, etc. Both an electronic clock for timing and a PCM analytical telemeter use the beacon as their transmitter.

Clock

Quartz crystals now being produced for crystal-stabilized oscillators will perform with an accuracy of 10^{-7} at frequencies around 1 mcps in a thermostabilized environment.

If set at launch, 6 months later the clock will be 1.6 seconds off, which means that comparison with absolute time is only required for accuracies better than 1.6 seconds. Resetting of the clock is desirable since it improves accuracy and need only be employed infrequently. It improves accuracy to the point that no real time correction is required for all experiments (Figure 3). Conveniently one chooses a 800 kc aged crystal in a sealed glass container heated by a coil to 60°C. A regulator maintains this temperature at very close range. This frequency is then divided by a chain of multivibrators to 400 kc, 50 kc, 10 kc, 500 cps, 400 cps, and 100 cps.

Both 50 kc and 10 kc are amplitude-modulated with the standard NASA time code (Figure 3). The code has a frame rate of one per second and uses 4 bits binary coded decimal for seconds, tens of seconds, minutes, tens of minutes, hours, tens of hours, days, tens of days, and hundreds of days. "Zero" corresponds to 2 millisecond pulses; "ones" to 6 millisecond pulses, zeros are interlaced so that an average to 100 pulses per second pulse rate results. The code is generated by a small computer which uses a magnetostrictive delay line as the temporary storage element for 156 bits. External drive is provided at an 800 kc rate and one computation is performed in 200 microseconds. Four flip-flops in the loop are used for the four bit binary coded decimal code generation. Additional logic determines when ten tenths of seconds and ten hundredths of seconds are reached and it drives two flip-flops to generate a 10 cps and 1 cps square wave signal. Appropriate circuitry converts zeros and ones as they occur in the time code to 2 millisecond and 6 millisecond pulses respectively as demanded by the code. The timing code then modulates the two coherent carriers, 50 kc and 10 kc, as indicated in Figure 3. The lower frequency of the two is radiated continuously through the beacon.

The availability of a minimum computer in the clock suggested itself to make use of certain logic components for the secure command system which naturally cannot be described here. Two receivers connected in parallel with fail-safe isolation circuitry to implement redundancy receive binary coded signals and feeds the command logic. The timing is arranged so that approximately 30 commands can be given on an average pass, up to 128 different commands can be transmitted in total. In case of clock failure, unsecure commands provide a minimum capability for purposes of analysis.

Telemetry

The importance of spacecraft performance evaluation during its mission has been stressed before. The telemetry installed aboard the spacecraft is designed to provide information on spacecraft performance as well as sensor information. Accuracies of test point data to be telemetered range from better than 1% to the determination of the mere presence of a signal. One must further take into account the current state of telemetry development and its likelihood of improvement, the possibility of increased or changed demands in the future, overall complexity and reliability, and ease of ground data readout and handling. Consideration of the problems to be faced, and the information desired gave impetus to these items collectively and individually, and resulted in standardization in the use of PCM for all major spacecraft systems at the GSFC. The present state of the art does not seem to permit accuracies better than what a 7 bit code entails within the severe environment involved and the over half a year longevity of the spacecraft. Many data points sampled or sensed are valuable only if the time when reported is known or the spacecraft's place in orbit is known. Naturally, the data rate varies greatly and some test points must be analyzed continuously; on others, one transmission per orbit is adequate. Those considerations lead to the conclusion that a tape recorder must be employed which at the same time determines at least one boundary for a maximum data rate. Conversely, a maximum data rate can be established recalling the fastest stability acquisition rate of .05 degree/second for spacecraft control system. A sampling rate of one per second seems to be adequate for this system.

Two independent PCM telemeters are provided, one to be recorded continuously on an endless loop recorder and the other to be commanded at any time. In the recorded telemeter, which shall be referred to as the "A" telemeter, a frame consists of 64 words and each word of 7 bits plus a word sync bit. The sync word is all ones and the word sync bit is a zero. Word number 33 to 48 is subcommutated into 16 columns so that 256 channels are available at a data rate of one per 16 seconds. The remaining 16 channels of the first row is subcommutated into 16 columns also having a sampling rate of one per 16 seconds, however, the arrangement was made that further subcommutation is possible. By adding an additional 256 gates, and making a small change in the timer, the number of subcommutated channels can be doubled at only half the sampling rate.

"A" System PCM Telemetry

Subcommutation identification is made by the second word that carrier the column number. By excluding the 256 channel gates, the "A" system is limited to 542 channels. It follows that a 500 pulse per second bit rate is required; this is supplied by the master clock. In case of failure of the clock, a tuning fork oscillator replaces the clock by unencoded ground command. A coherent 500 cps subcarrier is modulated by the coder output and recorded on an endless loop recorder. The (240 ft) tape passes the single record/playback head at 0.4 inches per second.

"B" System PCM Telemetry

The "B" telemeter cycles through 128 data channels preceded by three synchronization words, the first one being all ones succeeded by a word all zeros and another word containing all ones. The bit rate of 10 pulses per second is again furnished by the master clock. The pulse train resulting from coded output uses the non-return-to-zero method. A coherent 5000 cps subcarrier signal derived from the master clock is phase-shift keyed by the pulse train. When commanded for "B" telemetry the 10 kc time code signal is turned off so that interference is eliminated and a higher modulation degree can be used on the transmitter.

Command Playback

A playback of the "A" telemeter is performed separately by command. Power is applied to the playback motor which drives the tape through an appropriate drive mechanism at 12 ips, thirty times faster than the record speed. The 500 cps subcarrier is now converted to 15 kc covering a spectral bandwidth from very low frequency components up to 30 kc. Any one of the three signals — time, telemeter "A" or telemeter "B" — modulates the 350 milliwatt beacon-transmitter to 80% of amplitude.

Since the weakest links in the subsystem are the tape recorders and beacons, by ground command, the redundant units can replace units that have failed so that evaluation of the failure is possible for as long as required. Furthermore, automatic circuitry is prone to failure and thereby degrades reliability. In order to appreciate fully the complexity of the largest PCM system* ever built we shall proceed to review briefly a logic diagram.

*Produced by Radiations, Inc., MELBOURNE, FLA.

A 500 pps bit rate signal is provided by the master clock and converted to word rate in a shift register. There are six additional 16 bit shift registers; four are driven in sequence and the remaining two form a matrix with two registers driven in parallel, as shown in Figure 4. Timing logic provides a pulse for the first position of the first shift register; the second pulse, corresponding to the second word, triggers the first position of the second shift register; the third and fourth word are fed to the first position of the 3rd and 4th shift register. The fifth word opens the 2nd position of the first shift register. The two pairs of shift registers form a 256 position matrix and are timed in the conventional fashion. Each position of the registers and the matrixes opens a gate corresponding to a particular channel. There are 542 inputs through 542 gates available as mentioned before. Parallel connection of all the gate outputs could be prone to failure because a single short would eliminate all channels. For this reason a number of isolation gates are provided so that failure of a single gate deactivates a limited number of channels. The time sharing multiplexer, as described, feeds a single connection to the coder, which contains the analog to digital converter, a serial to parallel converter, and the synchronization generator. Analog to digital conversion is accomplished by comparing the input signal to binary weighed reference voltage applied through gates at a faster rate than the fixed bit rate. For power conservation coding is performed within 50 microseconds at a rate of 200 kc. The code is then stored in a core buffer where the word sync bit is added and read out at the bit rate of 500 pps. Additional windings on the cores are used to apply the sync word for the subcommutation code generator. Frame sync is generated by application of maximum voltage into the A/D converter. A return to zero output from the coder converted to non-return-to-zero and this gates a single cycle per bit of the 500 cps coherent signal which in turn drives the record head of the tape recorder through a record amplifier.

The logic for the "B" telemeter is very similar to the one used in the "A" unit. The multiplexer samples the 128 channels and feeds them through isolation gates to the coder. A parallel to series converter uses the 10 bit/second master clock signal for readout in a similar fashion to the "A" system. Presence of a second identical coder permits its redundant use in case of failure in either unit. Transfer is accomplished by command (not shown in Figure 4).

Justification for the use of the particular modulation techniques and a listing of system parameters is in order. The choice of AM on the transmitter was made first to satisfy the minitrack requirement for a stable carrier with no sidebands of significant power closer than 1000 cps and second for convenience of autotrack acquisition. AM further lends itself to most simple modulator circuitry. The 350 mw cw power is achievable with transistors rather than vacuum tubes so that a reduction in power possible by application of wide band modulation techniques would not substantially change the circuit design concept, but would complicate the modulator. Subcarriers must be used because of the dc nature of PCM and because of the carrier to sideband separation needed. Phase shift keying is easily accomplished in a ring modulator and will yield a small modulation improvement.

A more detailed discussion of the properties of tape recorders will follow and it may suffice to say at this time that speed variations of the tape result in phase and frequency variations so that a detector for a phase shift keyed signal from a tape recorder would be difficult to design. By recording amplitude the amplitude instability of magnetic tape is traded against the phase instability, however tests proved it to be the lesser evil.

Transmitter Power

The choice of transmitter power is determined by the following consideration:

1. Telemetry must be received even when the satellite tumbles for analysis of control system check points.

2. Allowance for a high fade-margin should be made since the spacecraft is larger than one wavelength of the frequencies involved and design of an omni-directional antenna pattern is very weight consuming.

3. Marginal reception at the horizon or under tumbling conditions is permissible.

Using the range equation:

$$\frac{P_{\text{received}}}{P_{\text{transmitted}}} = \left(\frac{\lambda}{4\pi \text{ distance}}\right)^2 \quad \text{antenna gain product}$$

the path attenuation of 146.5 db is found by using the assigned tracking frequency of 136.5 mc and the horizon distance of 3700 km. Because of the possibility of the spacecraft tumbling a requirement was introduced that we must attempt to generate a turnstile pattern on the spacecraft antenna. Such an antenna is ideally consisted of two dipoles arranged ar right angles to each other and each is fed in quadrature. Because of the right angle mounting is experienced no coupling and each can be considered independent of the other. One finds that radiation in the antenna plane is linearly polarized, normal to the direction of propagation radiation normal to this plane is right-handed circularly-polarized in one direction, and left-handed in the other; propagation in any other direction experiences an elliptically polarized signal. Receiving equipment must always experience a loss which will never be greater than one half (-3 db). Ground receiving antennas are 85 foot parabolic dishes with low noise parametric preamplifiers, driven by separate cross polarized linear elements. The antenna gain is 26 db at 136.5 mc. For computation of the signal to noise ratio the sky background must be taken into account.* Extensive measurements performed by Balton, Westfold, and Reber show that the antenna temperature will rarely exceed 1000°K. The receiver temperature can be neglected compared to this figure an one finds readily

$$-148 \text{ dbw} = \text{Noise Power}$$

using 60 kcps receiver bandwidth. By radiating 350 mw of power the ratio power received/noise power becomes 24 db so that an 80 percent modulated AM transmitter will yield 19 db S/N after demodulation. For a single bit error probability of 10^{-3} one would still find a fade margin of 10 db for imperfections of the antenna pattern. As a discussion of the antenna system will reveal, later reception will be marginal indeed for a tumbling satellite at horizon distance. However, for zenith altitude, 19 db are realized so that data can safely be received.

Arrangement of the channels is made so that 128 of the most important test points are telemetered by the "B" telemeter in addition to being recorded. The extremely low data rate requires only 100 cps bandwidth for the subcarrier so that a subcarrier predetection S/N ratio of about 45 db is available. A wide fade margin can be accommodated and phase shift keying will further yield a small modulation improvement so that under all conditions a negligible error rate will be experienced.

The spacecraft is now complete - except for an antenna system which will be explained later - and ready for service. This basic spacecraft weighs 500 pounds including antennas and consumes 100 watts continuously.

*H. C. Ko, "The Distribution of Cosmic Radio Background Radiation," Proc. IRE 1958, pp 208 - 215

TELEVISION SUBSYSTEM

Ever since the days of TIROS I launch and the reception of television, and the resultant insight into weather formations over the oceans where observation is sparse, the desire for complete earth TV coverage and rapid transmission to a Data Center became evident. Many considerations enter into the choice of system parameters such as linear resolution, maximum transmission time per pass, bandwidth, orbit characteristic etc., conversely time and bandwidth are interchangeable but time depends on the choice of satellite antenna beamwidth and so does its gain.

The resolution of a television camera and the postulation of area coverage and gray scale levels determines the information content. Since a satellite utilizing TV cameras of this kind can be considered an indefinite source of information, processing must occur at the same rate at which information is gathered. The time lag between collecting the data and receiving of the data at the ground station is an arbitrary choice but must not exceed a limit where its value diminishes. For instance, a weather-analysis map may be made from television pictures received at the ground station; that is, the weather must be forecast rather than analyzed after it happens. For this reason pictures are taken, stored in the satellite and read out when it passes over a ground station.

Ideally a single ground station should be placed at the north and south pole, but adverse conditions at those locations make this impractical. A survey of more conveniently located sites quickly reveals that more than one station is needed if every pass is to be read out. Even by placing some burden on the satellite in providing storage for two orbits, more than one station is needed if the true arctic station is to be avoided. Therefore, temporarily, only one ground station located at Fairbanks, Alaska is available (see Fig. 5). This station will be provided with a high-gain antenna (85-ft. parabolic dish) to intercept 10 passes per day out of 14 possibilities.

The question then arises whether to use a narrow-beam, high-gain antenna on the satellite or a wide-angle transmission cone. Rigorous treatment leads to mathematically trivial solutions because of the square law relationship of cone angle and antenna gain and the $1/R^2$ dependence on transmitter power. One can illustrate the problem simply by assuming an overhead pass and plotting the transition time of the antenna cone against the value of the cone angle, as shown in Figure 6. Conveniently, orbital altitude is used as a parameter. Since time and bandwidth can be exchanged linearly, transmitter power increases (or decreases) proportionally. Considering small antenna angles, one is led to conclude that the highest antenna gain and widest bandwidth (shortest transmission time) is the best choice so that equipment limitations would force a compromise. The plots further show that for the large angles a very favorable tradeoff of transmission time as against cone angle can be effected, thereby compensating for the inverse square law dependence on distance. Thus, clearly the last point shown on the graphs will be chosen. This is the earth disc-viewing angle one would see from the satellite vantage point. For a 1000-km orbit, this angle becomes 122 degrees with a corresponding time of 17 minutes. Orbital passes other than overhead will shorten this time; as a result, 10 minutes was chosen as a compromise.

The number of picture elements varies quadratically with the linear (or angular) resolution desired. Experience gained in the TIROS series showed that coverage is more important than resolution and that resolution elements need not be smaller than 1.5 km. Selection of television tubes becomes simple if circuit complexity, ruggedness, weight, and power are compared among the various choices. The advantage in sensitivity offered by the image orthicon, compared to the vidicon, is not sufficient for use in the earth umbra under star illumination; during daylight a vidicon is completely adequate. Angular resolution limitations are determined by the number of lines an electronic scan system can provide. Laboratory models of 1-inch diameter tubes with 1/2-inch photosensitive area have achieved 1200 to 1500 lines indicating that 800 lines is a reliable, practical limitation. Further related limitations of a different nature will become apparent from the tape recorder design characteristic that will be used as the storage medium.

Referring to the description of the retrograde high noon orbit, let us consider the geometry (Fig. 7) or such an orbit on the 21 of June when the sun is over the earth's equator. Suppose the satellite passes just overhead; after one period (107 min) it will again pass the equator. The earth rotates 27 degrees around its axis during that time (or an arc of 3000 km). Viewed from the satellite, the TV camera optics must encompass an angle of 108 degrees in order to view the area given by the 27° earth sector. This view divided by any multiple (M) of 800 lines yields the angular resolution and the number of cameras (M) required. Three cameras arranged like a fan (Fig. 7) would encompass 36 degrees along the equator from the satellite with a lens having a field of view of approximately 51 degrees. Linear resolution varies from less than one km (0.8 km) at the optical axis of the center camera to more than 2 km at the corners of the side cameras.

In order to determine the optic characteristic fully one must derive the light-energy level for the photosensitive surface of the vidicon. This depends on the camera's sensitivity and possible exposure time. Exposure times of finite value cause smear of the picture when the camera moves as demonstrated by a photograph from a camera held by a human hand. Camera movement referred to earth surface is 6 km/sec, so that smear less than 10% of a picture element demands that exposure time must be held to less than 130 milliseconds. The residual instability of the spacecraft is held to ± .05 degrees/second and a similar computation for less than 10% smear leads to exposure times less than 90 milliseconds. In the worst case the errors will add linearly; by allowing a small safety factor, 40 milliseconds exposure time was decided on. Basic lens parameters are now fully determined; however, search for a commercial product has not revealed a lens with these exact values. A Bell and Howell 17mm lens with a 49 degrees field of view was finally selected as being close to those angles and exhibiting acceptable optical properties combined with rugged design and small weight. It will be noticed that illumination will decrease with increasing northern or southern latitude, and that the three-picture pattern overlaps from orbit to orbit at positions other than at the equator. Rigorous elimination of the redundant information thus generated could save as much as 30%; however, turning the two side cameras off when the center camera could handle the swath width yields only a saving of very few frames, so that no attempt is made to make use of information reduction.

Observation of Fig. 7 reveals that illumination decreases from points north or south of the equator. Previous measurements and knowledge of cloud properties permits us to determine an f number of 16 over the equator and of 4 near the poles. This latter setting corresponds to a sun angle of 5 degrees, a value experimentally determined with the TIROS I satellite by taking pictures in succession over a night-dawn-daylight section of the earth. The optic contains a variable iris which is continuously varied from an f-16 setting to an f-4 setting according to Lambert's law by a cosine potentiometer attached to the solar paddle shaft and a motor drive.

A suitable timer programs the camera and tape recorder operation so that pictures are taken every 108 sec, exactly the time the satellite needs to transverse from picture-center to picture-center. One finds readily that 32 three-picture sets cover the illuminated half of one orbit from a near-south pole to a near-north pole. The storage medium then must accommodate more than 64 frames. For switching the system, the cosine potentiometer generating cosine voltage variation for iris setting over the orbit feeds a switch with an adjustable threshold. When the voltage is below this threshold, the cameras are turned off. Since no more than 10 minutes will be used for one series of 64 frames, 9.4 seconds are available per frame. Since the resolution has been selected as well as the time from antenna and orbital considerations, the video bandwidth is easily found. In such an idealized computation the maximum video frequency is generated when the beam scans a pattern where one half of all picture elements are white and the adjacent elements are black. One further property of the vidicon tube offers flexibility to the system designer: the image on the tube illuminated the photoconductor which in conjunction with a dielectric generates a charge pattern, thus certain storage properties exist in the tube. A considerable advantage becomes evident since exposure time of the tube and scan time can be quite different.

As exposure must be limited for stability reasons to 40 milliseconds, by time considerations we found that approximately 9.4 seconds would be available per frame on transmission, and further that no more than 108 seconds can be used from one frame to the next. Use of the various times for frequency conversion or expansion demand different record or playback located on each side of the 18 boxes mechanically actuating the blinds so that an opening is assured corresponding to a 25°C. nominal temperature.

Testing of the spacecraft and calibration of the sensors will not be discussed in this presentation. However, many design features although have been influenced by the testing requirement. Flight spacecraft will be tested at 7 g's r.m.s. white noise (gaussian) extending from 20 cps to 2000 cps. Approximately 30 days will be spent by the spacecraft in a thermal vacuum test, half of the time at 45°C and half at +5°C. This represents a margin of 10°C beyond the maximum error of the thermostat.

Calibration is performed by mounting appropriate calibration equipment with test patterns or other targets in an adapter section connecting the spacecraft with the rocket carrier. This procedure, more accurately termed a "check of calibration", since primary calibration is performed prior to installation of the system in the spacecraft, must be performed in vacuum as will be described later. It consumes approximately 30 days in a space simulator chamber possessing cryogenic and high vacuum capabilities.

Power Supply

The power supply is a solar conversion type that has been designed as a separate unit, but is closely associated with the design of both structure and controls. Solar power has been chosen for best weight economy and because it has sufficient reliability and longevity to be used for extended periods of time in a space environment.

When the Nimbus spacecraft is orbiting at 1000 km in a circular orbit, it will spend 69 minutes in sun light and 38 minutes in the earth umbra. Referring to Figure 1 it can be deduced that 41.040 cm^2 of silicon cell surface area is available to intercept solar energy corresponding to 5.7 K watts during sun illumination. The fraction that is finally available during the entire orbit is subject to a large number of parameters which must be taken into account. Efficiency of silicone cells can be selected to be better than 11 percent for a large production yield even when 2 by 2 cm cells* are produced. However, if a more detailed analysis is desired the efficiency degradation is due to increasing surface temperatures, various types of filters used to reject certain radiation heating effects, and space environment erosion.

The various subsystems require a multiplicity of voltages with many conflicting voltage and stability demands. The only practical solution to solving these problems is in the employment of DC to DC

*International Rectifier Corporation under contract to RCA before the limit is reached, the speed is reversed and the three camera signals are processed through a doubler and mixer circuit with local oscillators, generating the frequency-sharing spectrum as shown on the schematic (Fig. 15). This information is summed with other to feed a 5-watt S-band transmitter. The center frequency is held stable to 10^{-5} by using a crystal discriminator in a feedback loop. A deviation of $\pm$ 1.5 mc is accomplished by the composite voltage using a veractor diode modulator. While the operational importance of television was mentioned before, operation over a half-year is virtually impossible to guarantee. Random component failures could cause incomplete orbital picture coverage. In order to increase the chance of satisfactory operation, a duplicate set of cameras, tape recorder and transmitter is included. Appropriate switching by command will eliminate defective components and replace them with redundant ones. Before an analysis of the FM/FM system, the remaining part of the spectrum associated with the lower half of the block diagram will be discussed.

HIGH RESOLUTION INFRARED SCANNER

Television will show the meteorologist complete cloud cover over the light side of the globe, measured at local noon due to the noon-midnight orbit. The midnight half of the orbit cannot be used because the vidicons lack sufficient sensitivity to televise under star illumination. No other tube is presently capable of the required sensitivity. As known from fundamental physics, a body at a certain temperature radiates energy having a spectral distribution which can be computed after Planck's law. In the case of the sun, the peak of emission takes place in the visible spectrum; this light, absorbed by the earth, in the case of clouds is reflected, making clouds visible from space. Conversely, because the earth is a body at 250-260 degrees K, the peak of its emissive spectrum lies at 8-12 microns. Clouds located between a detector and the earth (which is a radiator now, not a reflector) will shield the "light source" because they are at much colder temperatures. Considering atmospheric constituents and their effect on infrared radiation, one concludes that radiation measured in the 3-4 micron or 8-12 micron band will give a cloud-cover picture during night. TIROS II demonstrated this for the first time for the 8-12 micron band.* Infrared systems may choose from a large variety of detectors with many orders of magnitude difference in sensitivity among them. Time constant, noise properties, and variation of those with temperature, cover a considerable range so that a great number of variables must be taken into account. The much lower energy available from the earth at 4 microns can utilize lead-selenide as a detector with higher sensitivity and lower noise than the detector for the 8-12 microns radiation. Thus, a higher video output can be obtained from the weaker radiation than from the stronger (8-12 micron).

The infra-red detectors have design parameters for optical resolution based on the detector dwell-time on the target. In contrast to television, no image is formed; the detector only integrates the energy received from the target. Composition of a picture is achieved by scanning a mirror so that the detector continuously sweeps from horizon through the sky until it starts at the horizon again. The optical axis prescribes a plane. The assembly, called the radiometer, is mounted on the satellite so that this plane is normal to the instantaneous velocity

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*Bandeen, W. R., Hanel, R. A., Stampfl, R. A., and Stroud, W. G.; Infra-red & Reflected Solar Radiation Measurements from the Tiros II Meteorological Satellite," Journal of Geophysical Research (to appear October, 1961).

vector. Fitting line after line thus scanned to a picture requires that Nimbus advance the width of one picture-element during the time it takes the mirror to scan one revolution. The optical angle is thus determined by this method of scan.

The Nimbus high-resolution infrared radiometer (HRIR)* is designed for an angle of view of 2.8 x 10^{-3} radians, 2360 cps video bandwidth. It scans at 2 resolutions per second 700 elements over the 122 degrees from horizon to horizon, thus achieving 2.8 km linear resolution.

Following standard practice, to avoid dc amplifiers and to be independent of detector bias stability, the light beam is mechanically chopped, and the ac signal is amplified and rectified so that a video bandwidth from dc to 2360 cps must be processed. Following one particular scan cycle from horizon to horizon, an analog signal will appear, declining rapidly when the sky is in the field of view to practically zero-output voltage. The hot spacecraft over more than a 180° angle will produce saturation in the amplifiers followed by a zero sky signal. During this sky-sweep time, a permanent magnet on the mirror axis triggers a gate and a multivibrator so that three pulses are generated, serving to synchronize ground equipment. Stability of the spacecraft enters directly into the problem of recomposing the picture, since scan lines are required to be adjacent to each other. The stability rate as stated before is 0.88 milliradians per second or clearly 0.44 milliradians per scan. Within a 2800-meter picture element, a 440-meter smear might degrade the resolution to 3.02 km. This is not strictly true since the stability rate is ±05 degrees/second and over the entire orbit 3.240 km optimum resolution will be experienced.

Referring again to Fig. 8, the HRIR radiometer output is shown to modulate a 10-kc voltage-controlled oscillator -2.5 kc. The sky level corresponds to 10 kc and the hottest signal deviates to 7.5 kc.

A tape recorder** almost identical in design to the television recorder records the signal at 3.75 inches per second. A four-track head combination is used similar to the TV camera recorder. One track receives the radiometer signal; another records the 10-kc timing signal from the master clock. When one tape reel is fully unwound, the movement is reversed and the signals are switched to the remaining two tracks. The recorder continues to record until the reel is empty again

*Designed and produced by International Tel & Tel Company, Ft. Wayne, Indiana under contract to NASA/GSFC.

**Designed and produced by RCA/AED under contract to NASA/GSFC.

and then is stopped by a limit switch. When interrogation is commanded at this position or any position between, the direction of tape movement is reversed and the speed increased eight-fold. All tracks are applied to four heads simultaneously and local oscillator and mixers generate a frequency-multiplexing spectrum, as the figure shows. Both television and HRIR tape recorders carry momentum compensation motors. Rotation of the moving parts in either tape recorder will generate a reaction movement which would tend to turn the spacecraft; compensation will eliminate movement and save control-system energy. Compensation for start-and-stop transients, or for the different moments of inertia for fully-wound tape reels as opposed to empty reels, is not fully accomplished. By coincidence, flutter and wow requirements for this subsystem are almost identical to the TV subsystem.

Design principles of FM/FM telemetry systems have been treated exhaustively in the literature* and it may suffice to repeat the most pertinent facts. The spectrum of a frequency-modulated carrier contains many side bands spaced in multiples of the modulating frequency (theoretically broad-band) on both sides of the carrier.

The signal-to-noise ratio at the output of an FM discriminator is improved over the ratio which would be obtained by AM transmission; this improvement is proportional to the deviation. Considering white noise as the disturbance of a FM transmission link, an ideal FM discriminator responding to frequency only, not to amplitude, yields noise amplitudes increasing proportionately with frequency in the familiar triangular noise spectrum in FM reception. A complex wave form produced by adding many subcarriers will therefore experience different channel S/N ratios, the worst for the highest-frequency channel and the best for the lowest-frequency channel. Because of this, the voltages of subcarriers are not chosen to be equal among each other but are weighted so that the higher-frequency channels cause more deviation on the RF carrier than the lower-frequency channels.

The number of filters used in the satellite and ground equipment warrant a brief statement about their effect on a frequency-modulated carrier. It can be shown** that distortion in the amplitude response of a filter has negligible influence on the FM signal as long as the distortion is symmetric. Unsymmetry causes an increase in the time

*Nicholas, M. H. and Rauch, L. L., "Radio Telemetry," J. W. Wiley and Sons, 1956.

**Kuepfmueller, Systemtheorie der elektrischen Nochrichten - Kuepfmueller, Uebertragering, 1952.

cons ant so that ansients will not be reproduced faithfully; phase distortion of the filters enters strongly and causes amplitude distortion of the demodulated waveform and group delay. This is a real problem in transmission characteristic of TV-type signals, since the relative time-scale within a sweep must be maintained. In contrast to all the disadvantages of FM, its advantages lie in the amplitude insensitivity of magnetic-tape recordings and the noise-improvement p. perties in both the S-band link and the subcarriers. It is noteworthy that cross-talk, the most bothersome effect in multiplexing, is largely rejected like any noise signal.

$$\left(\frac{S}{N}\right)_{rms} = \frac{\Delta F_{rf}\,\Delta F_{sc}}{2 f_{video}^{3/2}} \; \frac{\sqrt{B}}{\sqrt{\frac{1}{3} f_{sc} + \frac{1}{5} f_{video}}} \left(\frac{\text{Carrier}}{\text{Noise}}\right)_{rms}$$

ΔF_{sc} = peak subcarrier deviations due to video

B = i-f bandwidth

f_{video} = maximum video frequency

f_{sc} = subcarrier frequency

As explained before, an 85 foot dish antenna with separate linear elements is available with a preamplifier having a noise figure of approximately 4 db. For a 1700 mc signal and 3 mc i-f bandwidth a 25 db carrier/noise ratio is found since the antenna has 52 db gain.

Communication distance is again assumed to 3700 km to the horizon. Tumbling of the satellite need not be considered because television cannot be received from an unstabilized spacecraft. No polarization loss need be taken into account for the same reason. A small satellite antenna gain is realized but is not included in the computation because of the rather marginal 120° beam coverage as will be shown later.

By using the appropriate video bandwidth (60 kc or 2.5° K) and channel subcarrier oscillator frequencies and by performing this

*Baumunk, J. F. and Roth, S. H., "Pictorial Data Transmission from a Space Vehicle," Journal of SMPTE, January, 1960, pp 27-31.

calculation until total bandwidth requirements are met, individual S/N ratios in excess of 35 db can be achieved. Cross-talk and the noise properties of the magnetic tape will further reduce this value. On the other hand, the question arises as to what S/N ratio is required for visual presentation and what S/N ratio the detector yields (lead selenide or vidicon).

Commercial television can operate even under poor S/N ratio conditions as long as synchronization is maintained. Integration of the noise over many frames by the human eye averages the noise and adapts to the consistent parts of the picture. Since Nimbus pictures are a single frame only, the noise is frozen at the specific instant the picture is finally exposed on film. Extensive tests show that 20 db is desirable and even 15 db acceptable to the viewer. Noise in the vidicon is extremely low, and composite measurements of the vidicon and preamplifier show that the first stage determines the noise properties of the system. Careful selection of tubes yields 20 db ratios.

Taking into account detector noise, a lead selenide detector in the HRIR scanner still permits achievement of 35 db S/N ratio. However, deterioration results from the limited stability of the subcarrier oscillator. Still, the final S/N ratio exceeds 25 db. One could ask why telemetry power is not reduced or deviations decreased, thus saving bandwidth; the spacecraft will carry other equipment in the future which may need better S/N ratios, also it is desirable to let the detector, not the telemetry link, determine the quality of the measurement. The system is purposely not optimized, but overdesigned, adding safety margin for the long life required.

FIVE CHANNEL IR SUBSYSTEM (MRIR)

Experiments for Nimbus were selected at a time when no meteorological satellite had carried anything except television. The subsequent launching of TIROS II and III yielded measurements of terrestrial and reflected solar radiation, which were of a basic research* nature and had no immediate time-dependent application. For those reasons the same experiment flown in the TIROS series will be adapted for the Nimbus spacecraft. Measurements from a stabilized platform eliminate the difficulties encountered by those conducted from a spinning body such as TIROS. The choice of the optical bands is the same, namely:

1. 6 - 6.5 microns - water vapor absorption
2. 10 - 10.7 microns - atmospheric window
3. .55 - .75 micron - visible reference and daytime cloud cover
4. 7 - 30 microns - thermal radiation
5. .2 - 4 microns - reflected solar radiation

Channel 3 was chosen to give good contrast between clouds and background under sun illumination, to serve mainly as a reference and aid for the human mind. One can show that 99 percent of the back-scattered and reflected sun energy falls within the band of channel 5. Channel 4 covers the range of thermal emission of the earth. Albedo and thermal emission permit study of the energy budget of the earth. Channel 2 measures the temperature of the earth in a band where the atmosphere is transparent. Since clouds are generally cooler than the surface of the earth, a map showing isolines of radiant emittance can be interpreted as a cloud-cover map. As discussed before, this method is valuable since it works also on the dark side of the earth which is unobserved by television camers and can serve as a coarse backup for HRIR measurements. The difference between channel 4 and channel 2 is essentially radiation between 12 and 30 microns, characterized by strong absorption bands of carbon dioxide and water vapor. The spectrum of channel 1 corresponds to the region of water vapor absorption between 6 and 6.5 microns. The temperature profile and the relative humidity in the atmosphere determine the energy which can be observed by this channel. Although physically different the 5 channel radiometer uses the same principle as the HRIR radiometer, i.e. a scanning mirror with detectors mounted in the focal point. Again, in order to avoid d.c. amplifiers and for stability reasons in the detectors bias, the light beams are chopped and the signals amplified in tuned amplifiers. Synchronous detectors rectify the signal so that five quasi d.c. outputs are available. Resolution and other parameters were chosen to be compatible with existing TIROS

* Hanel, R. A. and Stroud, W. G., "Infrared Imaging from Satellite," Journal of the SMPTE, pp. 25 - 26, (January, 1960)

equipment. Following the same kind of reasoning as discussed for the HRIR radiometer, since a 50 km linear optimum resolution is desired, and 8 cps video bandwidth must be provided for a 2.85° field of view optic scan at 7.9 rpm. Again it is assumed that the maximum frequency is generated when half of the picture elements are hot and adjacent ones cold. The telemetry part of the instrumentation uses FM/FM, chosen mostly for historic reasons and because of ready availability of components than for any other reason. As the ground system will show, PCM would be a better technique but cannot be applied because of tape recorder limitations. The output of the five radiometer channels is fed to five subcarrier oscillators. These voltage controlled oscillators are of the phase shift type with symmetric amplifiers in the feedback loop, the gains of which are controlled by the balanced input signal. Each subcarrier oscillator is deviated 50 cps for full modulation.

The five frequency bands are:

100 - 150 cps
165 - 215 cps
230 - 280 cps
295 - 345 cps
360 - 410 cps

A 500 cps signal provided by the master clock serves as a timing reference similar to the 50 KC and 10 KC signal for TV and HRIR information respectively. For reasons of bandwidth limitations no time code is modulated on this reference frequency. The outputs from these six channels are summed and the resultant composite signal equalized for transfer characteristic correction in a record amplifier which drives the head of a miniature tape recorder. It is the same recorder used for PCM telemetry except for electronics. An oscillator provides an alternating current bias to the record head and the signal required for the erase head. For convenience, erase of the magnetic tape occurs immediately before recording. As had been discussed before the tape recorder is an endless loop, two-speed design running at 0.4 ips record and 12 ips playback speed.* The endless loop records continuously, day and night, except during a playback sequene. A hysteresis synchromous motor generates torque in the record mode through a mylar belt speed reduction. The motor is driven by the 100 cps two phase signal delivered by the master clock. Playback is initiated upon command by applying power to the playback motor and playback amplifier. This high speed motor is another 100 cps hystersis synchronous motor while a third motor provides momentum compensation. A low flutter and wow of 2.5% peak-to-peak measured without frequency limitations is achieved by using precision bearings and ground-in-place shafts having tolerances of better than 50 parts per million. A command pulse activates the playback motor, the playback amplifier, and a 2W 136 c FM telemetry transmitter feeding the antenna.

*Produced by Raymond Engineering Laboratory, Inc., Connecticut.

Calculation of the system parameters can be performed in a similar way as for the television system and due to the narrow band information to be transmitted, the resultant S/N ratio is excellent. For horizon distance (3700 km), the same ground antenna conditions and unity satellite antenna gain of a 90 kc IF. band width would show a 29 db predetection S/N ratio. Individual channel S/N ratios and their weighting could be computed using the same equation as for the Television system. Making use of the standard FM S/N improvement 35 db is the average S/N for the composite subcarrier oscillator signal.

In contrast to the cloud cover measurements, it is attempted to make an absolute measurement rather than the reproduction of a relative contrast. Accuracy is therefore determined by the stability of the oscillators, the tape recorder and primarily by the calibration of the instrument. Temperature stability of 1 cps has been achieved which corresponds to 2% absolute accuracy. The tape recorder speed ratio is subject to small variations as temperature varies. These will affect the absolute frequency of the five channels but can be compensated for by making use of the clock reference. The worst deterioration is introduced by the tape recorder. Speed variations such as flutter and wow generate frequency variations which appear as noise after discrimination. Flutter and wow up to 300 cps is less than 1.5% of the output frequency of each oscillator. It is only significant to measure up to 300 cps because higher frequencies are highly attenuated in the output lowpass filter. Related to deviation which is a measure for the dynamic range, a 10% peak to peak in-accuracy must be accepted. This clearly is the largest error in the entire subsystem. Accuracy would be only 20 db and the peak signal to rms noise ratio is approximately 30 db. Since accuracy and noise are determined by instrumentation properties and not by telemetry, no weighting of the subcarrier oscillators was attempted.

ANTENNAS

The problem of an antenna system for Nimbus has been introduced during previous discussions and desirable coverage requirements have been stated. A multiplicity of added constraints must be met for a variety of system considerations. Flexibility gained by separating controls and power from the remaining equipment in the spacecraft is further enhanced if it is possible to place antennas on the sensory torus and not on the interconnecting truss or controls. Mechanical interference with the shroud to be installed over the spacecraft during launch must be avoided. Finally the base area of the spacecraft should be unobstructed because the scanning sensors must have a clear field of view; since further measurements of the earth's atmosphere can only be conducted by viewing it from the base area so that a minimum of this area need be used for antenna. An attempt to design including all these constraints can only be made with the aid of experiments.

For radiation of any frequency selected between 1700 and 1710 mc, a cavity backed double spiral* radiates a circularly polarized wave. A fiber glass cone having the dimensions shown in Fig. 10 has two arms of two logarithmic spirals ($r = ke^{ay}$) of copper, these are preferably made like printed circuits. The 50 ohm feed point is located at the apex of the cone and the feed cable is carefully trimmed along one of the copper arms. The cone is mounted on one side of a cylindrical cavity. As the measured pattern demonstrates, a 110° angle was achieved at -4 db points, realizing a fair approximation to the desired 120° coverage. For implementation of redundant transmitters coupling measurements have been performed and as long as separation is held to 30" or more isolation will be 30 db or greater.

In order to have an efficient command subsystem for the spacecraft, a high power transmitter is used as the radiation source.

The most elementary type of antenna for this purpose is the whip or dipole antenna. An antenna of this type, when mounted on the top of the controls container, will show a null pattern in the axis of symmetry. Since the possibility that the spacecraft will be interrogated at the zenith is small, this restriction is not serious. A whip antenna shows the familiare circular dipole pattern in the plane normal to the axis of symmetry. Therefore when the spacecraft rises over the horizon the transmission ray encompasses 60° with the spacecraft axis of symmetry. It

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* Antenna system designed by Physical Science Laboratory, New Mexico State University, constructed by Missile and Space Vehicle Department, General Electric Company.

becomes necessary therefore to tilt the lobe maximum 30° toward the truss and maintain this shifted pattern regardless of the effect of solar paddle movement. The interrogation frequency is approximately 120 mc of a wave length of 2.5 meters. The solar paddles are 2.7 meters long, so severe coupling or reflection from the paddles must be expected. A series of scale measurements, see Figure 11, revealed a token amount of decoupling between paddles and whip if the spacecraft is given the electrical appearance of a cone. This leads to the employment of a conical mesh construction on the top of the controls housing, and a conical mesh skirt extending about two-thirds of the way down the truss structure from the controls housing.

Sample patterns taken with 0 degrees, 45 degrees, and 90 degrees rotation of the solar paddles reveal that a dipole pattern is approximated but that dependence on the paddle position is strong as the deep null for 0 degrees and 45 degrees paddle positions show. It shall be noted however that interrogation will hardly be commanded from this direction due to orbital geometry limitations.

By far the most difficult design is that of the 136 mc telemetry antenna. The maximum spacecraft height of 3 meters is quite close to the wavelength of 2.2 meters which is the wave length of this frequency. Consequently a turnstile pattern can only be approximated by using a multiplicity of radiators properly phased. This approach is rejected because it is extremely weight consuming and would require elements on the torus and other parts of the spacecraft. Selection of the radiating element itself fortunately meets with lesser difficulties. Radiating rods or slots are not considered because they are too bulky. The Quadraloop antenna*, developed by the New Mexico State University serves the purpose because it is the most compact, light weight radiator developed for the frequency (Fig. 12).

A brief description of the quadroloop antennas follows. A U-shaped conductor is filled with a suitable dielectric in the gap left by the U. Close to the shorted end of the U, drive power is applied across the two branches. The remaining part can be considered a transmission line. By loading the line at the open end with a capacitor the loop antenna can be physically shortened and tuned. Naturally those conditions hold for a rather narrow band only and tuning must be provided for the specific frequency. Match of the driving source is dependent on the distance from the shortened end and is experimentally determined.

The E vector propagates between the two branches of the U and generates a pattern essentially that of a dipole, the nulls being in the direction of the largest dimension of the loop. Four such elements are arranged around the sensory ring. Phasing of the radiators was experimentally determined and the reader must recognize that the unconventional phase relationship between the radiators has at present no reasonable

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* Haas, H. W. - Quadroloop Antennas, Final report NONR 2158 (01) Physical Science Laboratory, New Mexico State University.

explanation. It must be mentioned that the antenna beamwidth is within a very narrow 100 kc band, and the final pattern achieved is far from what the spacecraft system designer tried to obtain. Fig. 13 illustrates the patterns measured for these radiators. Two cross polarized linear receiving elements have been used and the field strength as represented by the signal output has been measured. The procedure was repeated for different solar paddle positions to show the effect of solar paddle rotation. For the case where the paddles are in the axis of symmetry there is a linear wave emitted towards the sub-satellite changing polarity and rotation towards the sides. Assuming that the spacecraft is stabilized, a good signal will be received even at 90 degrees (270 degrees) viewing angles towards the satellite.

The same conclusion is reached when the paddles are at 45° or 90°. In all three cones the upper half of the pattern is unimportant for the operating and functioning satellite. A circular receiving antenna suffices for this case. No polarization diversity is needed although the received power will vary approximately 5 db, in addition to the range variation. Suppose the satellite tumbles and the paddles are in an arbitrary position. If they are at 0° a - 11 db signal variation could be encountered at the tumbling rate and in addition, polarization changes from bottom to top will be noticed. The 45° and 90° paddle positions show even a more degraded pattern than that since a null develops near the top as the angle is increased. As the pattern shows no energy is emitted by either of the two linear components thus no power will be received and a periodic deep fade will be found. The pattern should be considered as being typical. A large number of pattern have been measured in order to verify repeatibility and the influence of the paddles. Due to the symmetrical shape of the spacecraft, with the exception of the paddles, a pattern shape is measured in the plane perpendicular to the axis of symmetry is more circular than the three examples shown.

In order to be prepared for the case of the tumbling satellite, receiving antennas must employ polarization diversity, for example, mutually perpendicular elements and separate receivers for each element. Even then tumbling would cause deep fades.

The five channel radiometer information is of no value when tumbling occurs and for this reason a certain directivity in the radiator associated with this subsystem is desirable. There is simply a single qualraloop, see Figure 14, mounted on the base section. The patterns resemble that of a non-ideal dipole, radiating energy towards earth and space and showing nulls at positions near the top. The pattern dependence on solar paddle positions are still present although much smaller than at the telemetry antenna. The similarity of the pattern between the telemetry antenna and the MRIR antenna and the difficulty which was experienced in changing it by trying different phasing or geometry lets one speculate that the whole spacecraft is the radiator and the elements merely exiters.

GROUND SYSTEMS

A spacecraft especially as large and complex as Nimbus is only part of a communication and data processing system. Considerations presented up to this point had to take into account certain ground equipment properties. These have been stated when needed. However, obviously the design of the spacecraft system must make use of ground equipment design limitations and it must be influenced by the final product. A great deal of thought can be given to a definition of what the final product shall be: perhaps a weather forecast to the local radio station, or to a ship at sea; a weather emergency warning when it is apropos; a weather chart for the forecasters use or merely a cloud map; or a mosaic of pictures compatible with standard map projections or unrectified versions of terrestial cloud cover and features. Similar to the fine distinction which lies between data processing and data analysis is the problem of data acquisition and presentation as distinguished from that of data utilization.

Data acquisition and presentation influence the system design strongly and in most cases are made part of the S/C system. Those and other considerations lead to the conclusion that standard film is the end product for picture type data and digital computer tape for the data serving further research. The picture data is applicable for TV and HRIR cloudcover, the taped data for experiments telemetering through PCM and for the 5 channel radiation experiment. It has been mentioned before that although incomplete orbital coverage is obtained unless a station is located at the north or south pole, more than one such source is needed for locations towards the equator. There is still only one station available for the Nimbus satellite program. This ground station is to be located near Fairbanks, Alaska, and will be used for the initial Nimbus spacecraft.

ANTENNA AND RECEIVERS

The 85 foot dish antenna at the ground station near Fairbanks, Alaska, contains separate cross polarized linear feeds for the 136 Mc band and 1700 Mc band, (See Figrue 16). Pointing of the dish requires movement around two axes which for mechanical reasons have been chosen to lie in the horizon plane. Such a drive system is called an X-Y mount in contrast to two axes driving azimuth and elevation respectively. Manual operation of such a narrow beam antenna is impracticable, thus automatic tracking modes will be provided. The antenna can be operated in three possible modes: (1) it can be driven by a tape recorder where aximuth and elevation data is stored as a function of time, (2) it can autotrack on 136 c, and (3) it can be operated in autotrack on 1700 c. To fully appreciate the pointing problem of such a large dish, it is interesting to note that the beamwidths derived from the antenna gains are 0.7^{o} for S-band and 10^{o} for VHF.

A typical pass will involve the following steps: (1) Computer predictions for time and azimuth will be made by the computing center and communicated to the station, taking into account the minimum elevation possible at that particular azimuth angle due to horizon obstructions. (2) A magnetic tape will then drive the dish at the programmed time. (3) Acquisition of the tracking signal is visually displayed and an operator can operate the mode of track switch to auto-track when the signal strength is sufficiently high. (4) When the S-band transmitter is turned on by the command system (radiated through a stack of disc-cones attached to the 85 foot dish), autotrack can be switched to S-band again, if the visual display indicates sufficient field strength for that mode of operation. The conventional monopulse system is used for tracking on both frequencies. After conversion of the 1700 mc signal to 137 mc and further to 30 mc phase detector comparators use a reference and generate separate X and Y error signals for drive of the dish. 136 mc reception uses the same conversion system though separate equipment. Note that separate parametric low-noise preamplifiers are used for both VHF and S-band vertical and horizontal antennas, making a total of four. Receiver* design was purposedly made rather universal because of the desirability of using antennas and receivers for satellites other than Nimbus, and for ground stations other than Fairbanks, Alaska. The two S-band preamplifier outputs are combined so that a circularly polarized wave is received then converted to 137 mc and fed to the multicoupler and discriminator. More than one frequency can be tuned for in this arrangement. The choice of more than one IF amplifier and second converter is available, although Nimbus uses only 3 mc bandwidth mentioned before.

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* "Antenna and Receiver System Design by Tracking and Data Systems Directorate," NASA/GSFC

The frequency discriminator feeds to a video low pass filter at approximately 800 kc roll off.

The VHF receivers are designed for the Minitrack telemetry band 136-137 mc which is used throughout the Minitrack network spreading from Alaska to Chile and stations in South Africa and Australia. Two low noise parametric preamplifiers, one for each feed, drive a monopulse system as described before. The horizontal and vertical signals are then fed to IF amplifiers which can be selected in the 30 kc, 60 kc, 100 kc, or 300 kc range as desired. Nimbus uses the 100 kc bandwidth for telemetry and a separate 100 kc IF receiver for reception of MRIR signals. The choice of an AM or FM detector can be made for each frequency. Nimbus will have its output connected to the AM detector of one receiver and FM discriminator for the second frequency. Diversity combiners though required for telemetry reception only, select the stronger of the two mutually perpendicular signals which are used for both 136 mc frequencies.

COMMAND CONSOLE

Communication with the satellite is established through the command system. (Fig. 16.) A series of commands is prepared by pushing a keyboard and punching the code on a paper tape. Every command is visually displayed and entered on the tape separately. When the preparation is complete the tape can be rewound and fed into a tape reader which may serve as a permanent printed record. The command sequence is now ready for transmission. This can be accomplished manually by feeding the tape derived signals to the transmitter modulator or be initiated at a certain time by the station clock. Provision is made to connect the keyboard to the transmitter modulator or be initiated at a certain time by the station clock. Provision is made to connect the keyboard to the transmitter by pressing the tape so that the operator control of the satellite can be assumed. Unencoded commands are tone frequencies transmitted at desired times by operator controlled switches. A crystal detector located near the transmitting antenna feeds its output to the same tape reader, printing a record of the actual signal transmitted in the proper time sequence. Visual comparison is used to determine whether or not errors occurred. Multiple use of the display units is made by utilizing them to record the time difference between satellite time and station time according to the synchronized WWV absolute time reference maintained at the station.

Reception of telemetry at the station is combined with a certain amount of data processing. The least number of errors occurs at the ground station, so as much data as possible is acquired and processed at this time. Transportation of records for analysis in laboratories is impractical where spacecraft attitude information, control system functions, the state of the spacecraft power supply, and spacecraft temperatures are being sampled by the telemetry system. Because of the need for this essential information in real time, and from orbit to orbit, decommutation, display, and printout equipments and receivers are provided at the data acquisition site. In addition, computers will also be provided to analyse the data gathered from the spacecraft.

Following the receiver output, the signal finds the three demodulators corresponding to the subcarrier oscillators. Time is fed to the command console for comparison with the station time. B telemetry is fed to a sync detector which establishes the presence of a signal and of bit sync making use of the trailing and leading edge of the sync pulse. Since the S/N ratio is good as we have found by analysis, the sync acquisition is good. When sync is acquired a decommutator is synchronized so that individual channel signals can be analysed if so desired. For most cases the serial code will be converted by a code converter and read out into a tape puncher for permanent record. The tape in turn can be fed to a conventional electric teleprinter to display channel number and measurement value. This small, simple minimum telemetry capability guides channel assignments in the spacecraft strongly because of the high reliability inherent in simple automatic equipment. If desired the tape can be transmitted to a control center for further analysis. As will be apparent later a second means for storage and display is available using components of the A-telemeter.

The 15 kc demodulator output of the A telemetry drives a sync detector for frame sync detection and for bit sync detection. The decommutator, decommutates all channels including subcommutation words and it is equipped with a patch-board programmer allowing selection of individual channels. The serial pulse train is suppressed by a squelch until the first frame sync is acquired so that channel identification is positive. Each word is stored in a 7 bit shift register which is emptied in parallel into a CDC 160 A computer. A magnetic tape storage unit operates in conjunction with the CDC 160A computer stores each word in standard digital tape format. The computer inserts frame number received and word number in each frame thus serving a primary filing function. Analysis of individual channels over one orbit is simply achieved, with calibration functions being stored as sub-routines.

For many channels the signal stays within limits as long as no failure occurs. Those channels whose limits are important are programmed and specially identified. When the readout is complete, playback of the digital tape can be initiated and a Analex CDC 1612 high speed printer is driven by the CDC 160 A. These devices print at a rate of 1000 lines/min. Since one frame is recorded at 1/30 of a second on playback it is equivalent to one second in the record mode; there are 6420 frames per orbit with 60 words each. These 385,200 words would need 200 feet of paper from the printer and would take 45 minutes to complete the operation. It is quite evident that real time printout of all channels is impossible and furthermore it would be impossible to inspect them in real time. By printing only words which are out of specified limits the huge quantity of data becomes manageable. For much data the measurand must be correlated with orbital position which in turn can be accomplished only by knowing the precise time the data was acquired. In order to amplify this statement we must briefly consider a design detail of the satellite tape recorder. The endless tape loop emerges from the cartridge, passes a guide roller, an erase head for a.c. erase, a record/playback head, and a capstan drive shaft assembly. In this arrangement information is stored as long as possible, the first recorded approaching the erase head, the last recorded just leaving the record head. On playback the erase oscillator is turned off, and the speed increased thirty times but the direction of feed in not changed. Consequently at the time of playback initiation the only clean portion of tape lies between the erase and playback record head. Although this portion is played back first it is usually lost during acceleration of the drive system to full playback speed. It has been mentioned previously that an appropriate gear train activates a cam and switch arrangement to reset the whole system to record mode. Actually the tape is made slightly shorter than what the switch determines to be one revolution, allowing this clean section of tape to be played back near the end of the playback. Information appearing after this string of zeros, appears a second time on the record (i.e. at the beginning and at the end). The time of this clean section is known within a second since the time of transmission of the command to the satellite is controlled. The duration of the clean section is approximately two frames so it can be used to tag the last word preceding it. This time is inserted manually into the CDC 160 A and later on to the tape unit. Since word numbers and frame numbers are known, time of each event can be determined by the computer by counting in reverse.

In many instances a selected printout will not be the optimum form of data presentation, i.e., the charge-discharge cycle for one whole orbit is best presented in analog form, also certain temperatures need only a gross analog presentation. The patch board permits selection of up to 32 channels and connection to 32 digital to analog converters, the output of which is fed to a 32 channel galvanometer type recorder. Since the entire orbit appears highly compressed but less accurate, correlation of data can be established most rapidly.

Loss of telemetry data must be avoided, particularly those losses due to equipment breakdown. A duplicate computer digital tape storage device is included in the system, and guards against losses due to long down times in this unit. Furthermore the analog subcarriers are recoded on one channel of a 7 channel mincom 107 tape recorder. This affords additional protection against inadvertant loss of data. A second track records the output of the three subcarrier demodulators with a third track recording station time as put out by the station clock.

5 CHANNEL MRIR DATA PROCESSING

In agreement with the principles stated before, a digital tape is the end product for the 5 channel IR radiation data. Processing time is not a consideration because research depends primarily on ideas and not as much on time. Activity at the station for this reason is confined to recording the receiver output on one track of the same Mincom 107 recorder. As the experiment evolved historically out of the TIROS series, further data processing is performed for the Nimbus version using identical equipment. Analog tapes will be mailed to the Washington laboratories where the equipment is located,(see Fig. 17). When played back the 15 kc reference channel is frequency discriminated. This error signal adjusts the Mincom recorder speed so that 15 kc is approximated. The spectrum of channel 1 through five, i.e. from 3000 - 12,300 cps is transformed to 53 kc - 62.3 kc. Crystal filters demultiplex the signals, feeding five discriminators. A separate discriminator detecting the reference at 65 kc delivers the high frequency noise components due to fast speed variations in any of the tape recorders in the link. These are added, 180° reversed, through delay modules to the individual discriminator outputs thereby reducing this type of noise by approximately 1/3 or 10 db. An analog to digital converter and digital tape recorder converts the five analog signals sequentially to binary form. Computers can now assume the final chore to list radiation levels for each data point and its geographical position. For this orbital information and attitude information, documentation from telemetry must be available.

CLOUD COVER PRESENTATION EQUIPMENT

When this subject was introduced, the conclusion reached was that pictures shall be the end product and film the storage medium. Therefore, it is desirable to reach this goal with a minimum of intermediate steps. The choice is essentially limited to an electronic scanner or an electromechanical scanner.

The advantage of the electronic scanner is the very high scanning speed which naturally can be the same as the one used for vidicon readout. It needs, however, an intermediary transducer to transform the beam current variations to light by means of the phosphor illuminance which then can be photographed (i.e., an open camera integrates the light spot as it scans through the picture area). Faithful reproduction of a high number of quality gray levels is difficult to obtain by means of the phosphor, although it yields the same quality as the vidicon. Electronic scanners are subject to drift and their scan pattern must be readjusted frequently, say, prior to each pass.

Electromechanical scanners such as those used in galvanometer recorders have been in use for more than 40 years and produce a very stable scan pattern. They need adjustment much less frequently than the electronic scanners; a feature which is important for continuous operation at a near arctic field station. The dynamic range of light sources converting current variations to light variations is large and very linear. The disadvantage of these facsimile recorders is their relatively slow speed, which is certainly slower than the vidicon readout speed. Each of the two devices has its place in the Nimbus TV system. The Kinescope monitor, using the electronic scan principle, will be used for quick look evaluation and the electromechanical facsimile recorder for high quality reproduction.

The composit subcarrier oscillator signal as received from the output of the S-band discriminator is fed to a set of filters for demultiplexing. Local oscillators and mixers convert the signal to the doubled value of the original subcarrier oscillator frequency, i.e., 180 KC. The three TV signals and the time carrier are recorded on 4 tracks of a standard 14 track tape recorder using 1" magnetic tape. As the signal is received a monitor can be connected to any one of the three channels. It uses the Kinescope technique of converting the electrical signal to light variations on the phosphor of a cathode ray tube; the beam of which is deflected electronically. The light spot is photographed by a movie type camera. A control unit generates orbit number, date, camera designations (right, center or left), and time to the nearest second automatically, or by usual setting, as appropriate.

Illuminated numbers are photographed simultaneously with the picture presentation for this purpose. Once the film is exposed a rapid developer permits observation and quick look analysis shortly after exposure.

Pictures to serve as masters for further reproduction and research purposes are generated using the galvanometer scanning technique. As explained before the maximum speed of those is restricted so that bandwidth compression, or time expansion, must be used.

Most standard instrumentation tape recorders are designed for various speeds. The Mincom 114 will be used at a 60 ips record speed and played back at 7.5 ips. Because of the doubler in the satellite, playback video bandwidth is 15 kc while playback time has increased 8 fold so that 80 min are needed for an entire 2 orbit playback. As discussed before, double orbit recording is only needed once a day, so that normally 5 min. playback will suffice. In view of the 107 min orbital period, picture reproduction can be accomplished within one period. When played back the three TV tracks are fed to three discriminators which in turn drive facsimile recorders. These recorders function like mirror-galvanometer recorders. A film moves continuously at the line speed, generating the line advance. A galvanometer carries a mirror projecting a parallel light beam projected on the film. Constant current through the galvanometer coil will deflect the coil and mirror and sweep the light beam across the film. Similar to the movie sound recorders a light bulb is intensity modulated serving as the transducer from current variations to light variations. The properties of the film and those of the light bulb demand a non-linear intensity characteristic in the electronic drive amplifier for correction. Galvanometer speed can be estimated by assuming that retrace over the film width shall be accomplished within the time of 2 picture elements. It follows that the galvanometer cut off frequency must be above 3.75 Kc.

System characteristics discussed earlier show that speed must be constant to better than .1% (.01% is a desired limit). Since this value is very difficult to achieve and it remains that it can be maintained over 1/2 year of operation, provision was made to apply compensation. Speed variations will be reflected as a frequency variation on the 50 kc timing channel. By discriminating these, an error signal is generated that compensates for speed variation when superimposed on the sweep drive. Naturally a 180° inversion is needed. Location of a certain picture or part of a picture on the globe is only as accurate as the $\pm 1^{\circ}$ stabilization accuracy permits. In addition non-linearities in the optic and in the electronic scanning system degrade the accuracy further. This latter error is compensated for by measuring lens distortion and by etching fixed markers on the vidicon face plate. Corrections can then be applied by using prelaunch calibration pictures. 35 km uncertainty generated by the control system is generally unimportant for meteorology but the possibility of error determination exists within the limitations of the controls horizon sensor. By use of PCM telemetry, deviations can be computed at the ground station or at a different location at a later time. It is interesting to note that the narrow band time expanded signal is the most economical for transmission over radio links whether microwave or scatter propagation techniques are used. A control unit similar to the one used for the monitor generates indexing information for exposure of the film, thus keeping a log as complete as possible.

Picture reproduction of the night time cloud cover is accomplished in a manner similar to that used for the TV system. A facsimile recorder and index control unit is used to compute indexing information and the design is almost identical to the corresponding units in the TV. Since the picture strip is continuous unless interrupted by switching to the second track or by activation of the night-day switch in the satellite, an index will be applied at the beginning of each strip using the presence of the first time code as a trigger signal.

The most important difference between this system and the TV recorder is its ability to correct for errors in the control system. One may recall that the rate of change of all three axes of the spacecraft is held to ± 0.05 degrees per second which in turn causes 1.6% smear per picture element. This error is small and need not be corrected. Assuming, however, that larger errors may be in existance due to partial equipment malfunction or that future satellites may demand higher resolution, a correction capability is desirable, thus its basic elements are included in the recorder. Correction of control errors in the direction of the scan sweep is not necessary because the sync pulses initiate the sweep and the scan rate is much higher than the rate of stabilization. Tape recorder errors are corrected in the same fashion as in the TV system. Adjustment of the film speed is necessary to compensate for the errors of the tape recorder and the same drive can include errors of the control system in this axis. In addition, one must recall that drift of the orbit as a function of time generated by the difference between the desired retrograde drift and the real one introduced by guiding errors or altitude errors at injection, cause a deviation from the desired 90° between the velocity vector and optical axis. This can be corrected by cocking the galvanometer sweep against the normal film movement. Such adjustment is best performed manually for each orbit and held constant since the rate of change is measured in the order of days. Control System errors causing the same effect can be corrected in the same fashion, although this must be done continuously. Since the HRIR scanner is not a momentary exposure of an entire image but rather a scanner where the entire vehicle participates in the image generation, the location of each picture element is only as accurate as the pointing accuracy of the spacecraft, since 1° is much larger than the 2.8×10^{-3} rate. The field of view, and knowledge of where the elements lie is restricted to this accuracy. The complexity of servo mechanisms does not enhance trouble-free operation in the field, particularly since error correction must be generated by the attitude information stored in the CDC computer, and must be in synchronism with the ground sweep.

The facsimile recorder contains all the servo drives required for real time readout of nighttime cloud cover pictures. However, their application is not planned until further experience is gained with the spacecraft or it becomes possible to increase resolution of the detecting instruments. Positioning errors due to the $\pm 1^{o}$ control system error corresponds to ± 17.5 km on the globe, this is acceptable for meteorological analysis as discussed for the TV data presentation. This is especially important since the system designer must include reliability considerations either resulting in nighttime cloud pictures or some other method of presentation.

System design per definition involves signals from its sensing to the delivery of the end product. The real end product lies in meteorological analysis and only engineering considerations, ones ability to produce equipment at certain times, and ones knowledge of how much real information is in the measurement, has determined what constitutes the Nimbus satellite system. In the future we will modify this system concept as one will have learned more about the desired end product. Those future system designs will have to answer the same question; namely what to transmit and how shall it be presented.

Acknowledgement: Many details of the subsystem presented here have been designed by engineers of the NASA/GSFC. Participation of the companies mentioned in the footnotes in the implementation of the Nimbus system is greatfully acknowledged. System integration and test is performed by the Missile and Space Vehicle Department (MSVD) of the General Electric Company, Philadelphia, Pennsylvania.

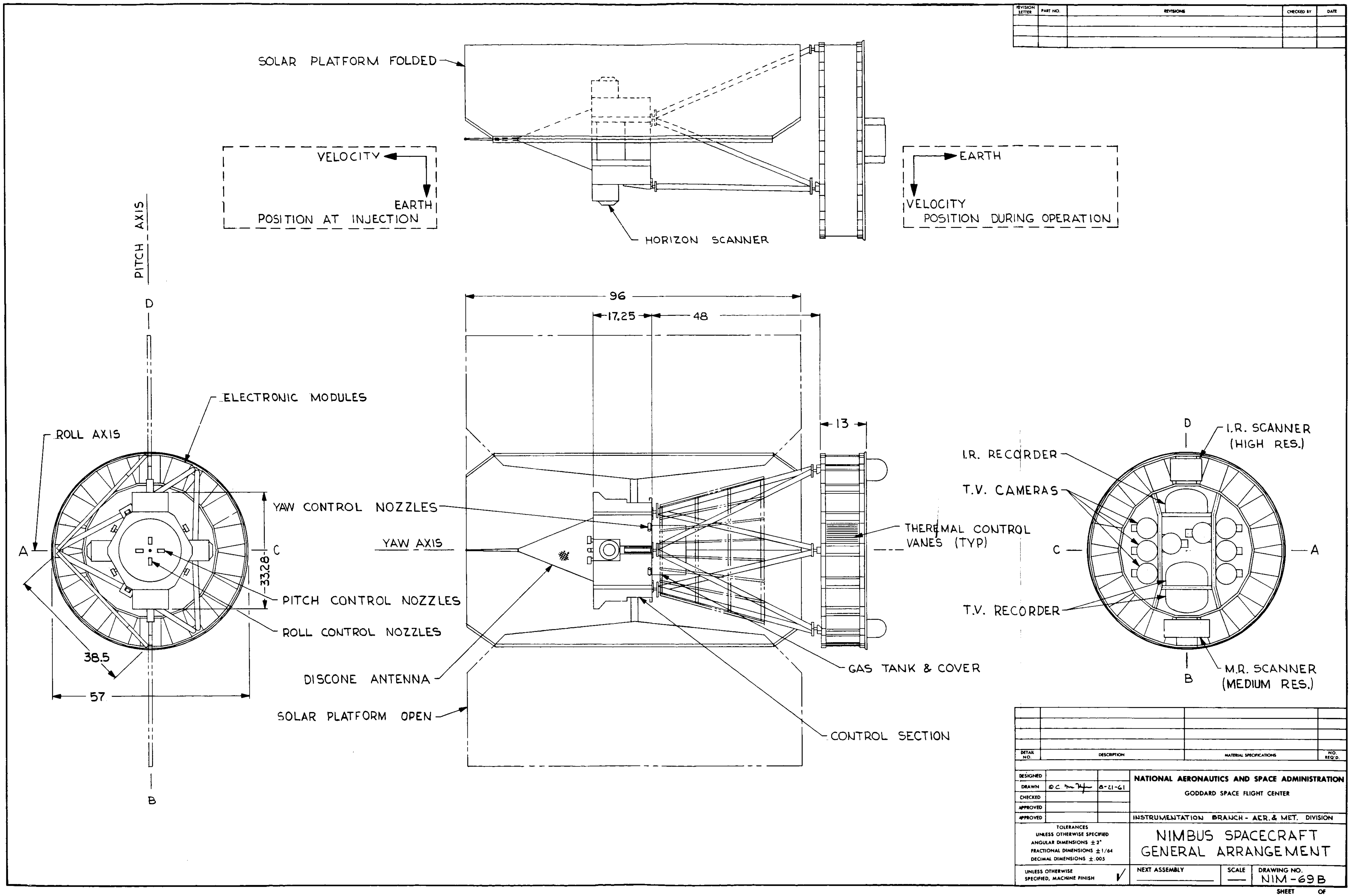


Figure 1. Nimbus Spacecraft

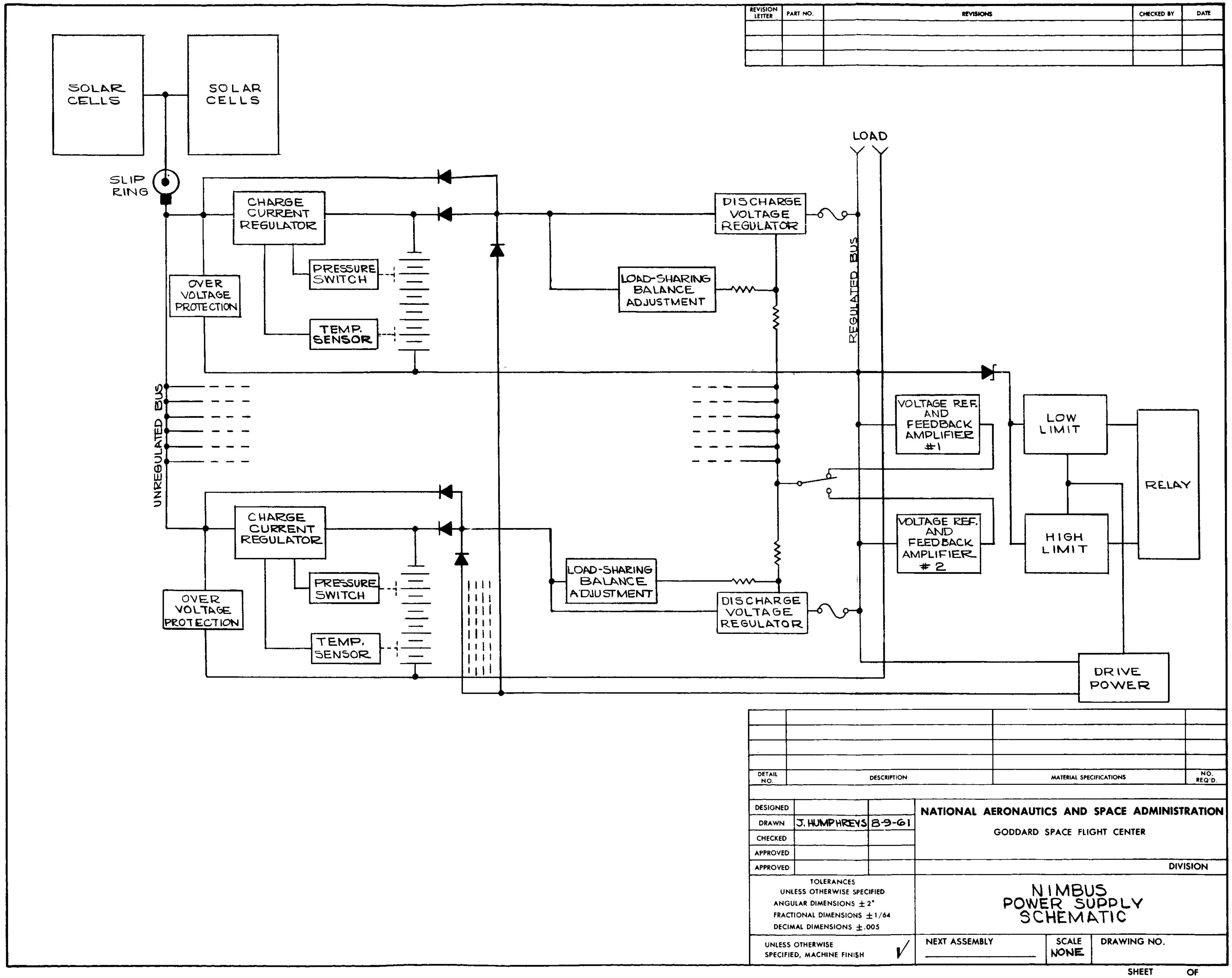


Figure 2. Nimbus Power Supply Schematic

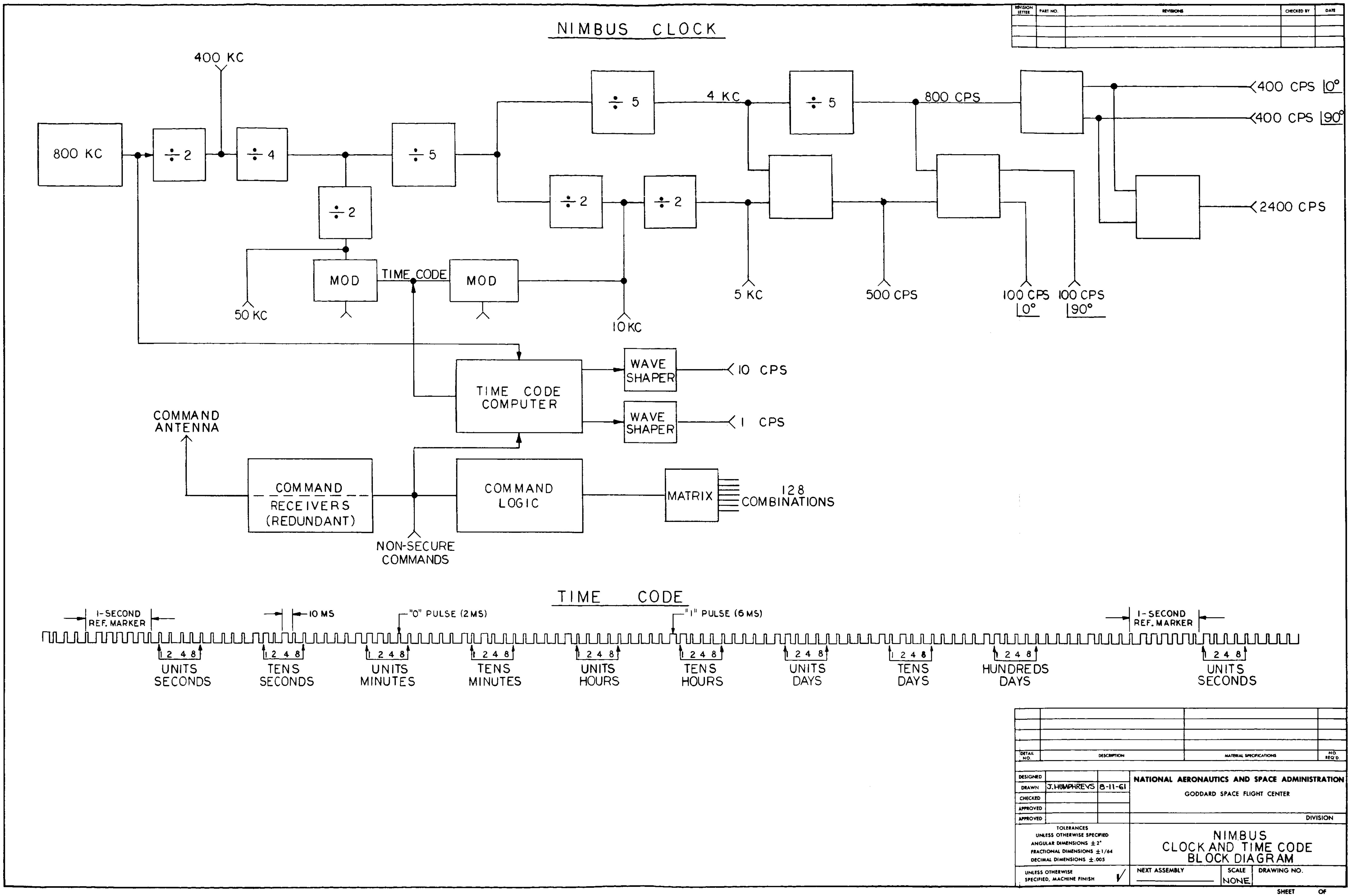


Figure 3. Nimbus Clock

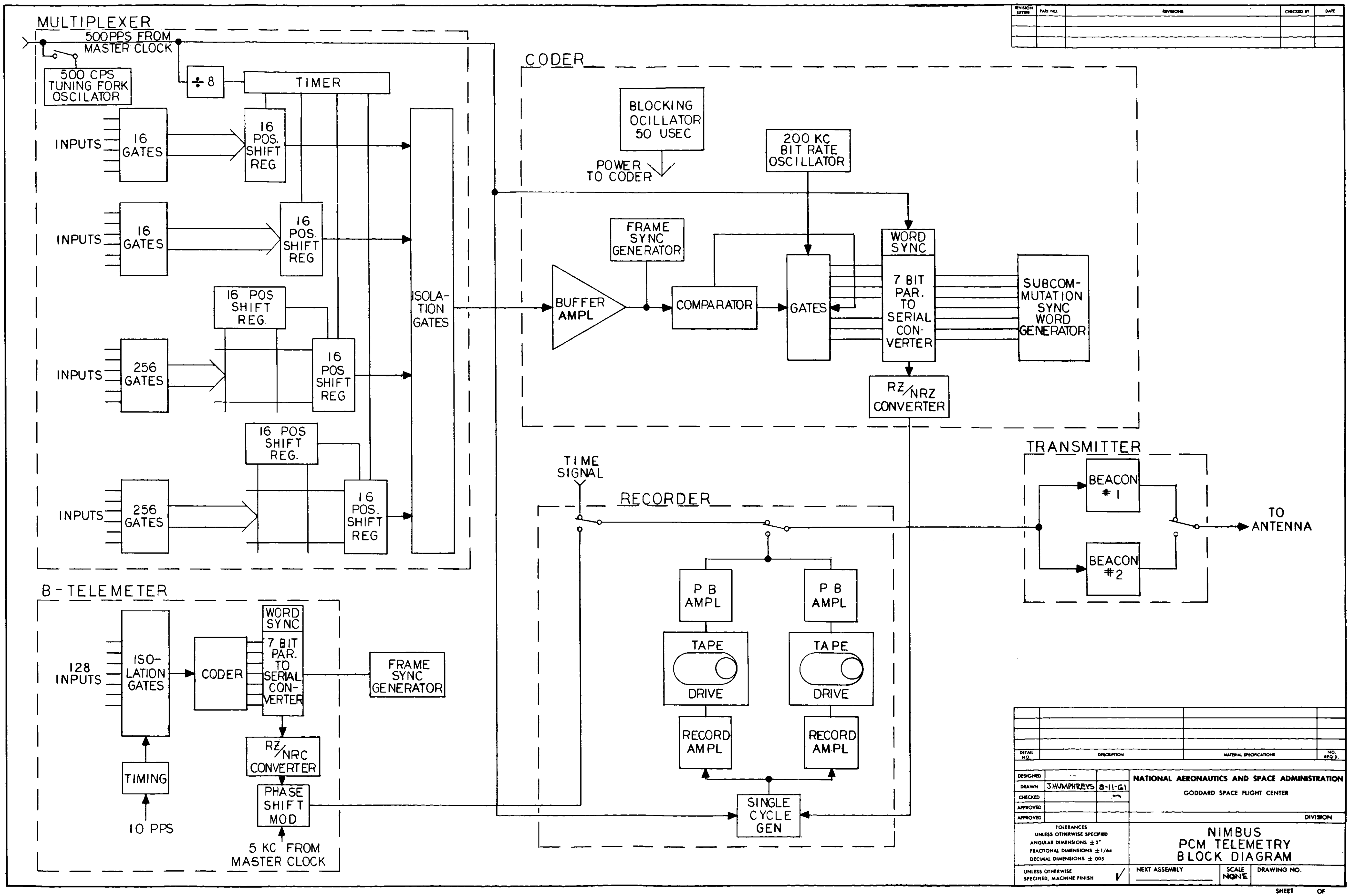


Figure 4. Nimbus PCM Telemetry

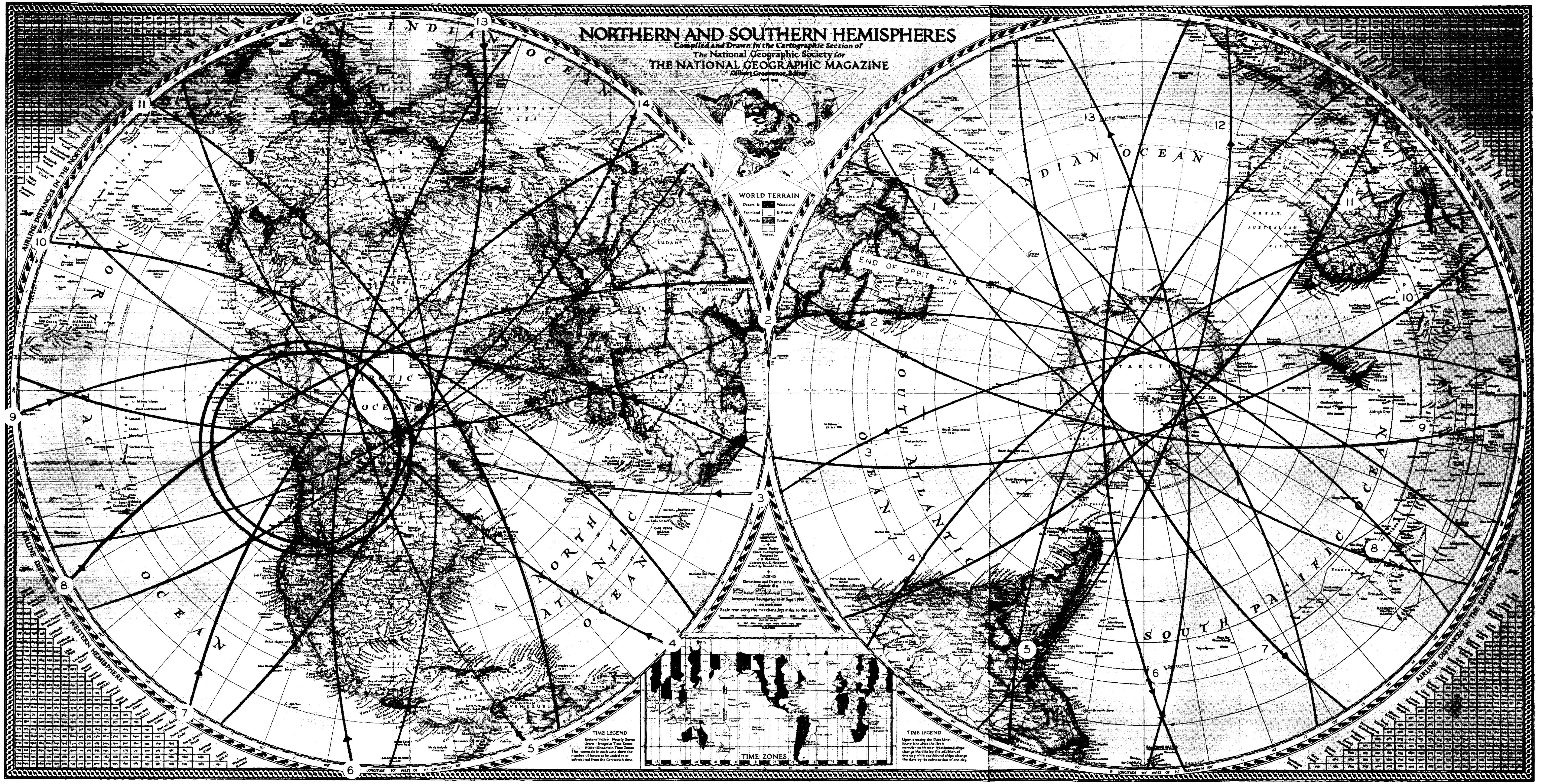
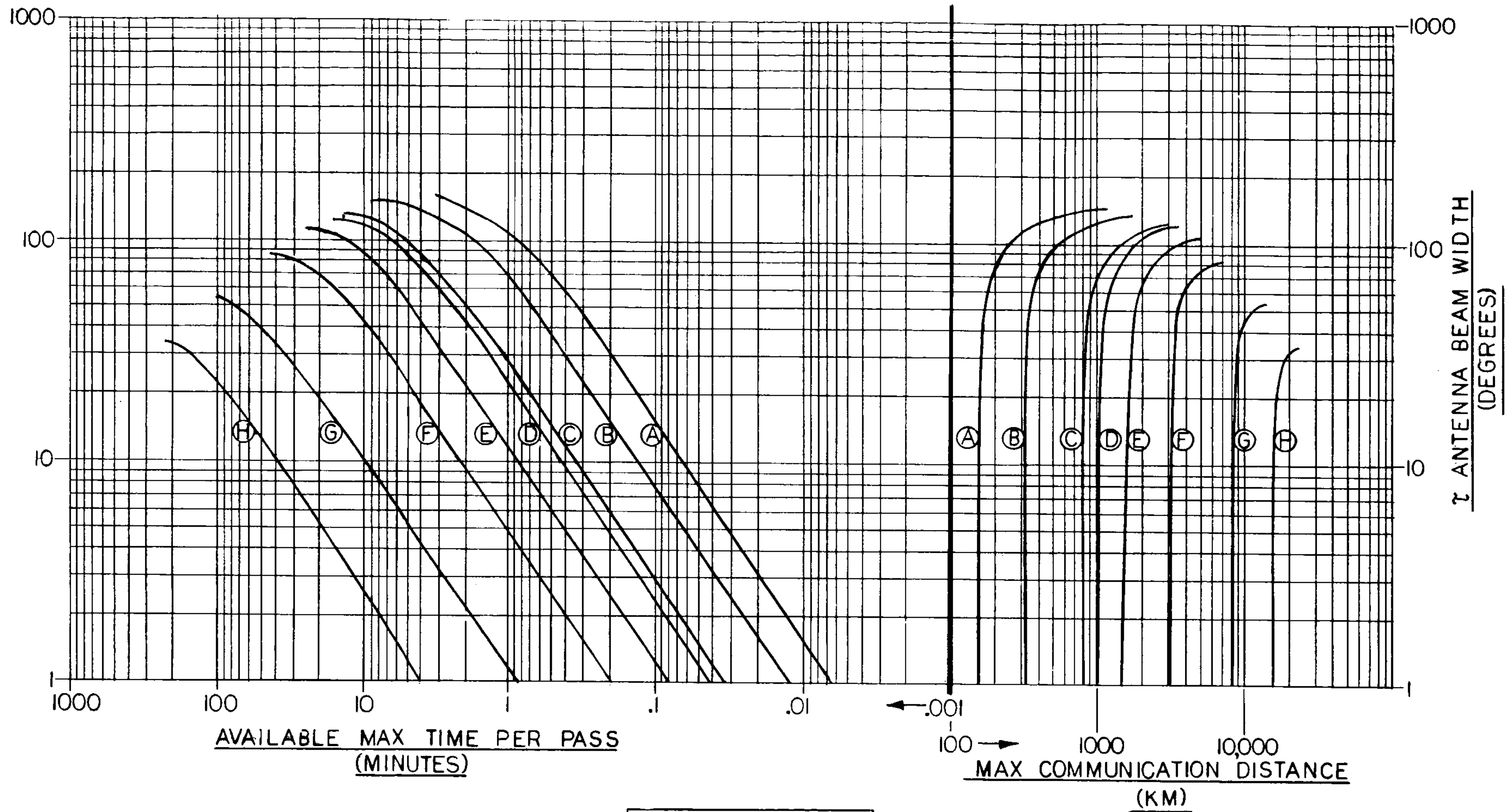


Figure 5. Orbital Coverage from Fairbanks, Alaska



HEIGHTS		
SYM	KM	MILES
A	160.9	100
B	321.8	200
C	804.5	500
D	965.4	600
E	1,609.0	1,000
F	3,218.0	2,000
G	8,045.0	5,000
H	16,090.0	10,000

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Figure 6. Antenna Beam-width, Telemetry Time and Communication distance

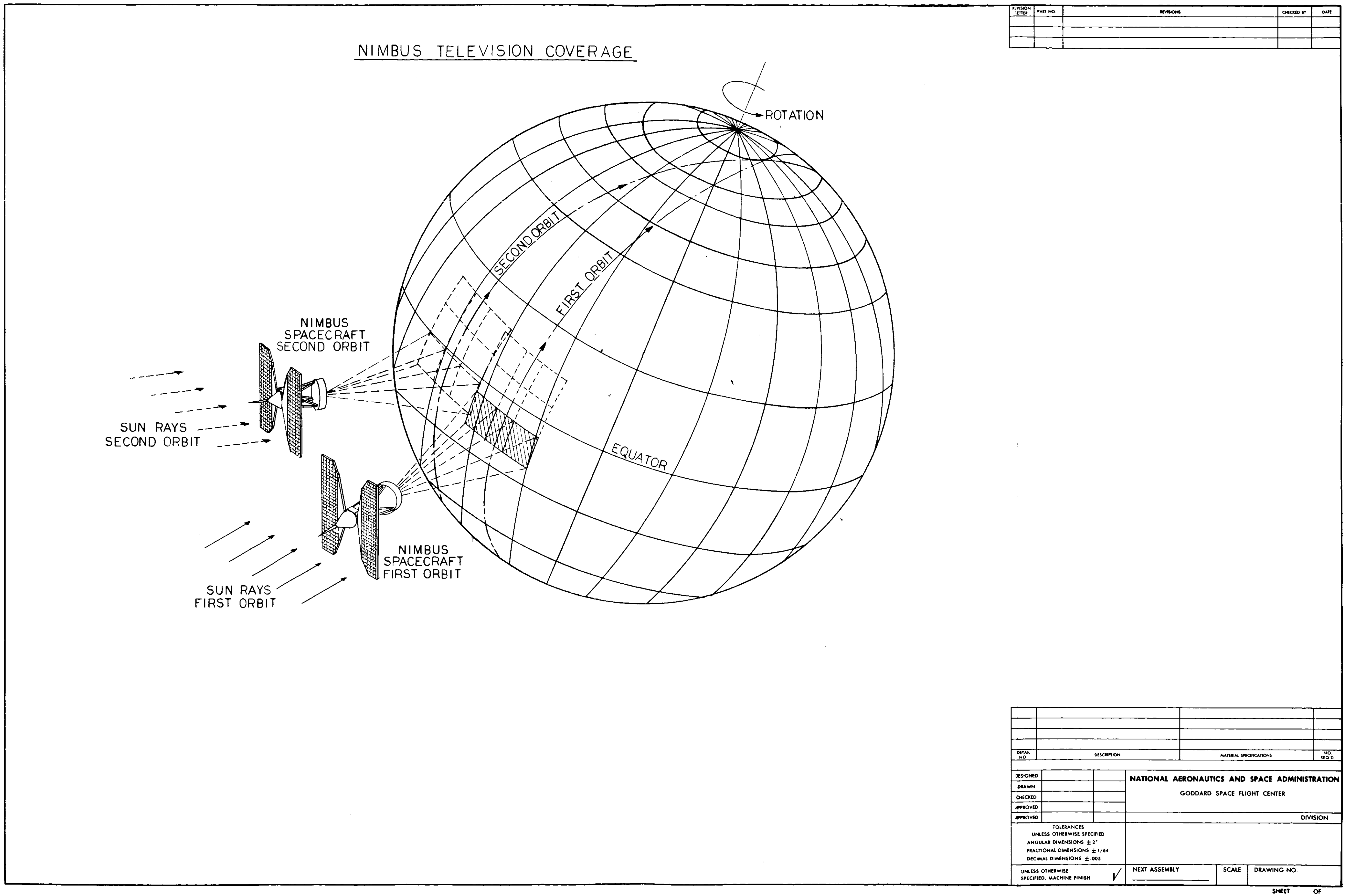


Figure 7. Nimbus Television Coverage

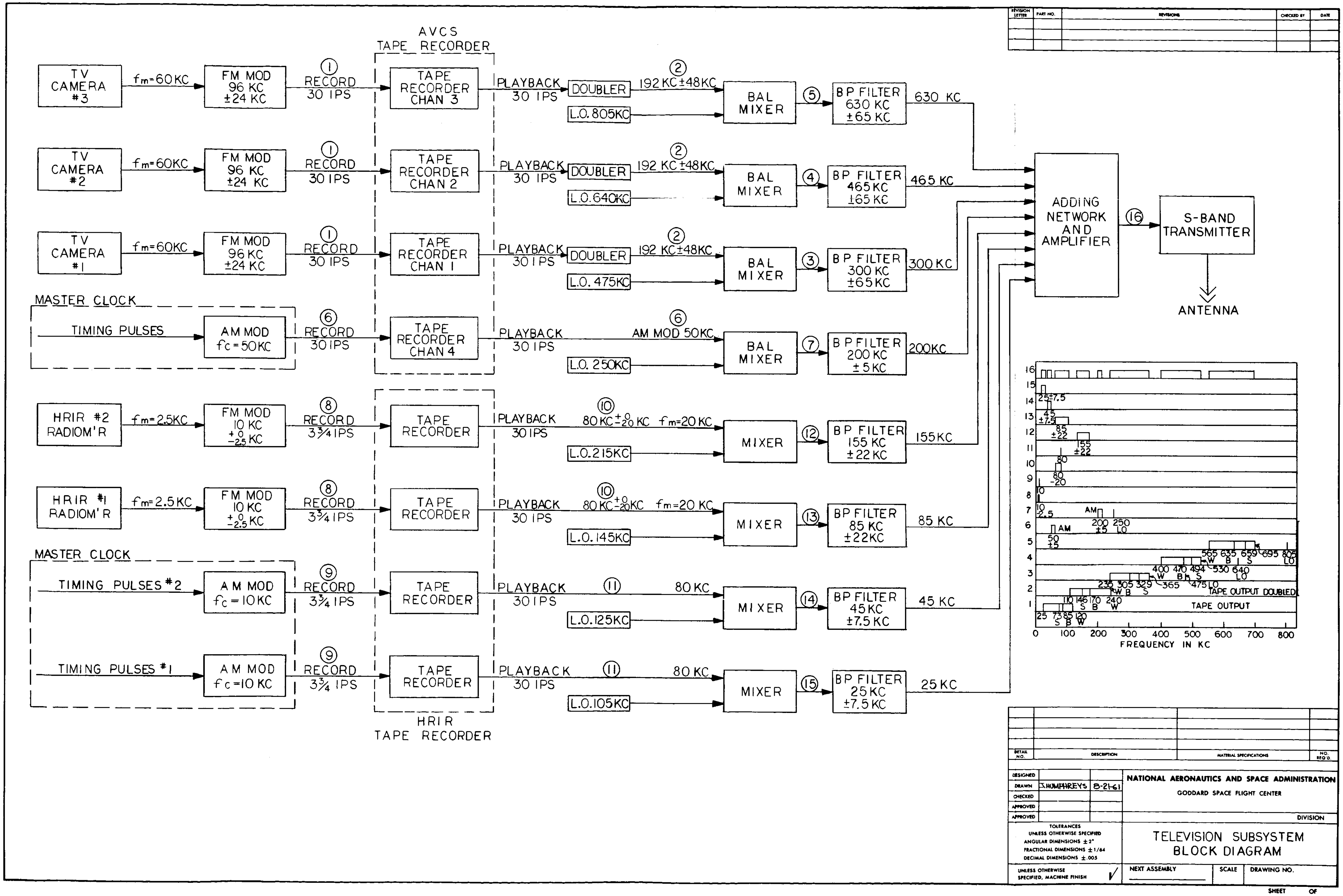


Figure 8. Television Subsystem

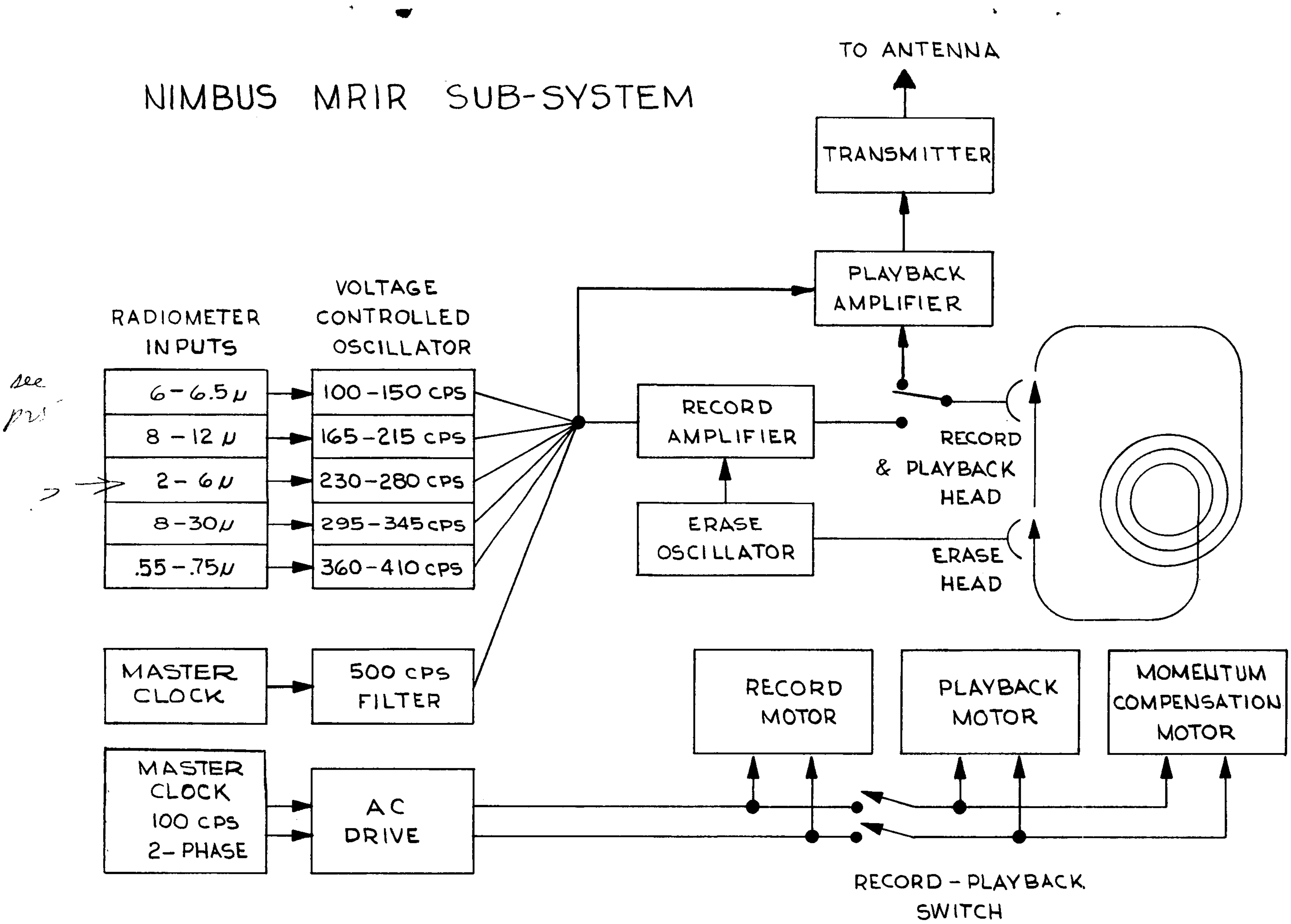


Figure 9. Nimbus MRIR Subsystem

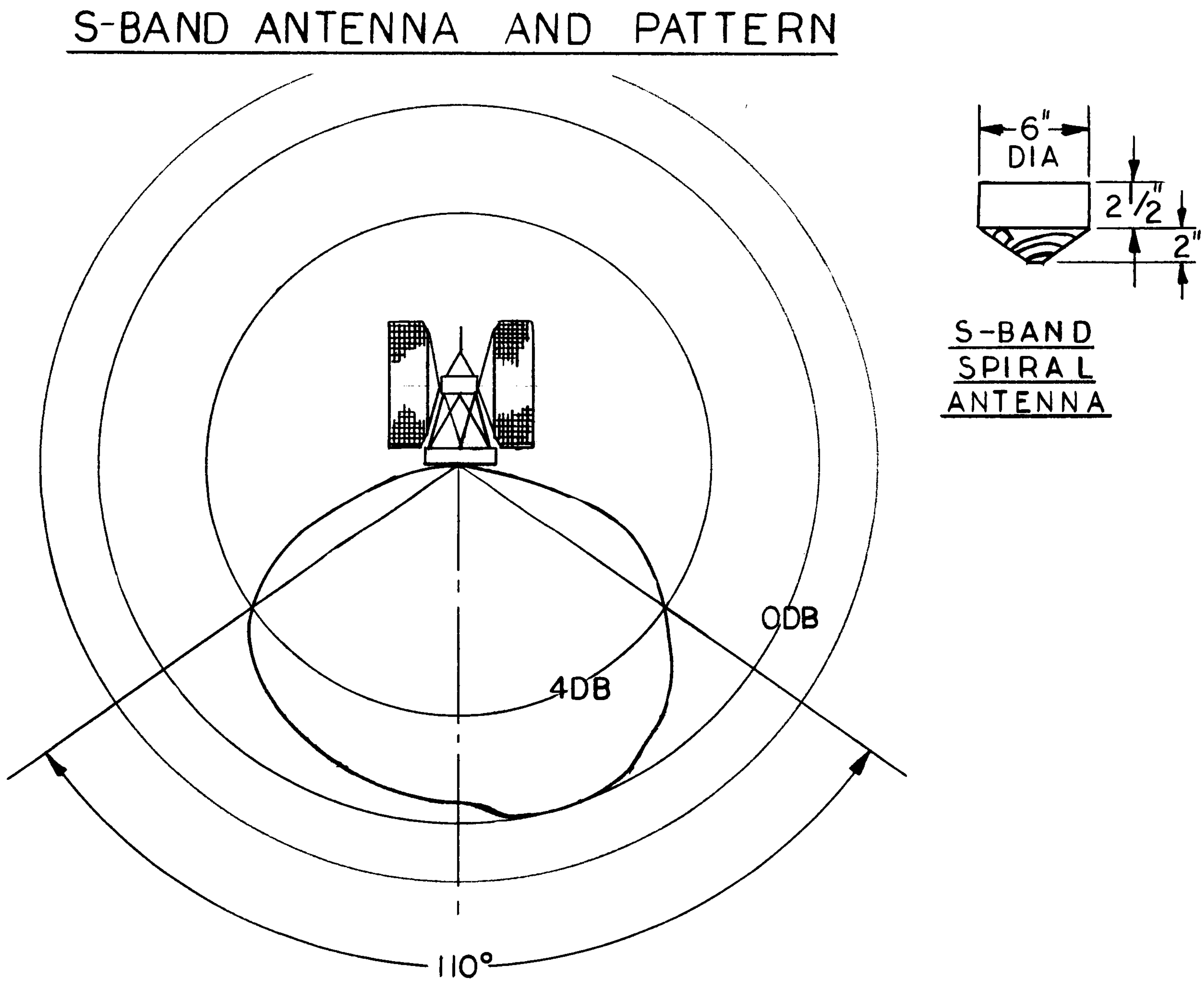


Figure 10. S-band Antenna and Pattern

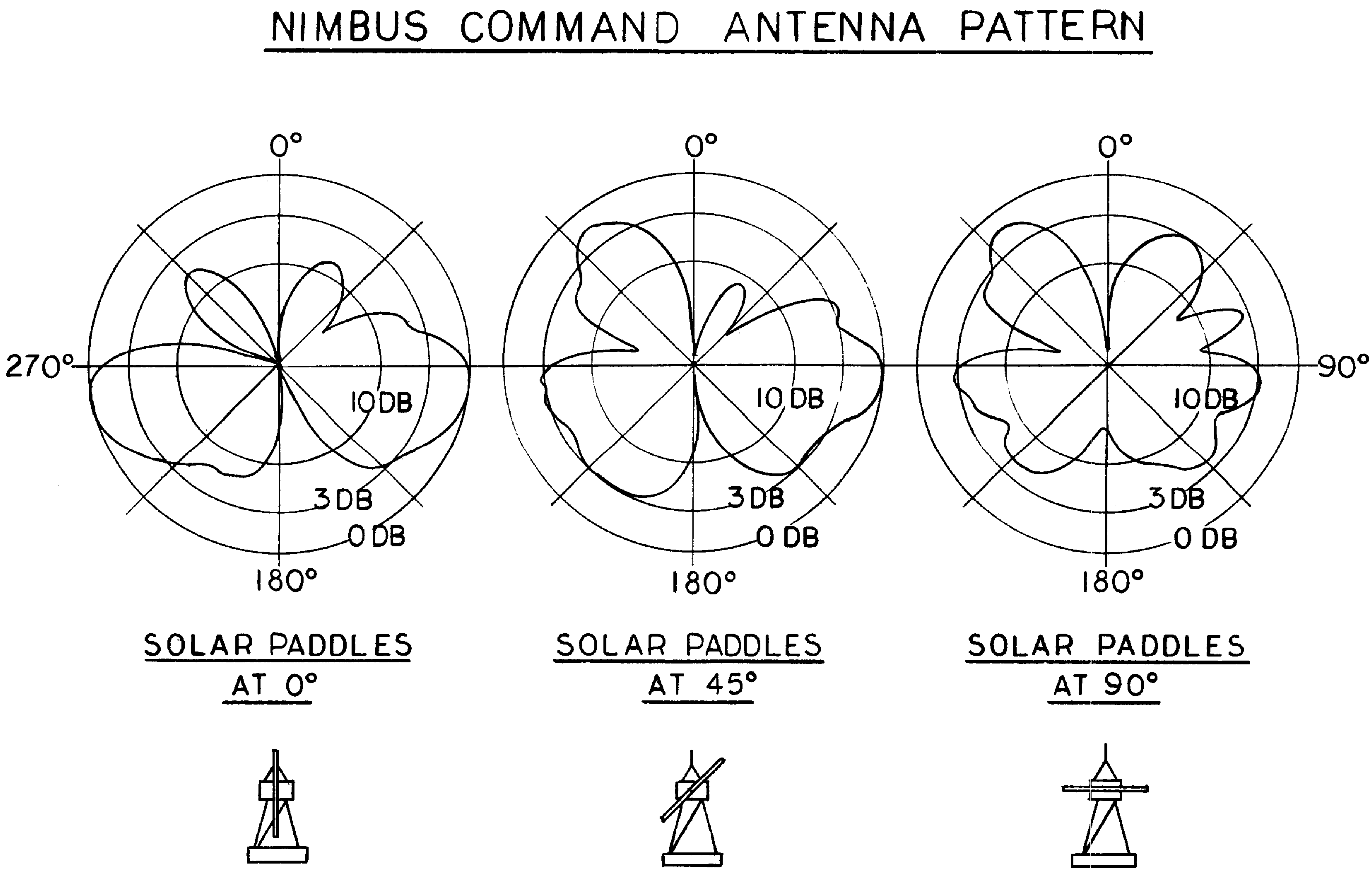


Figure 11. Nimbus Command Antenna Pattern

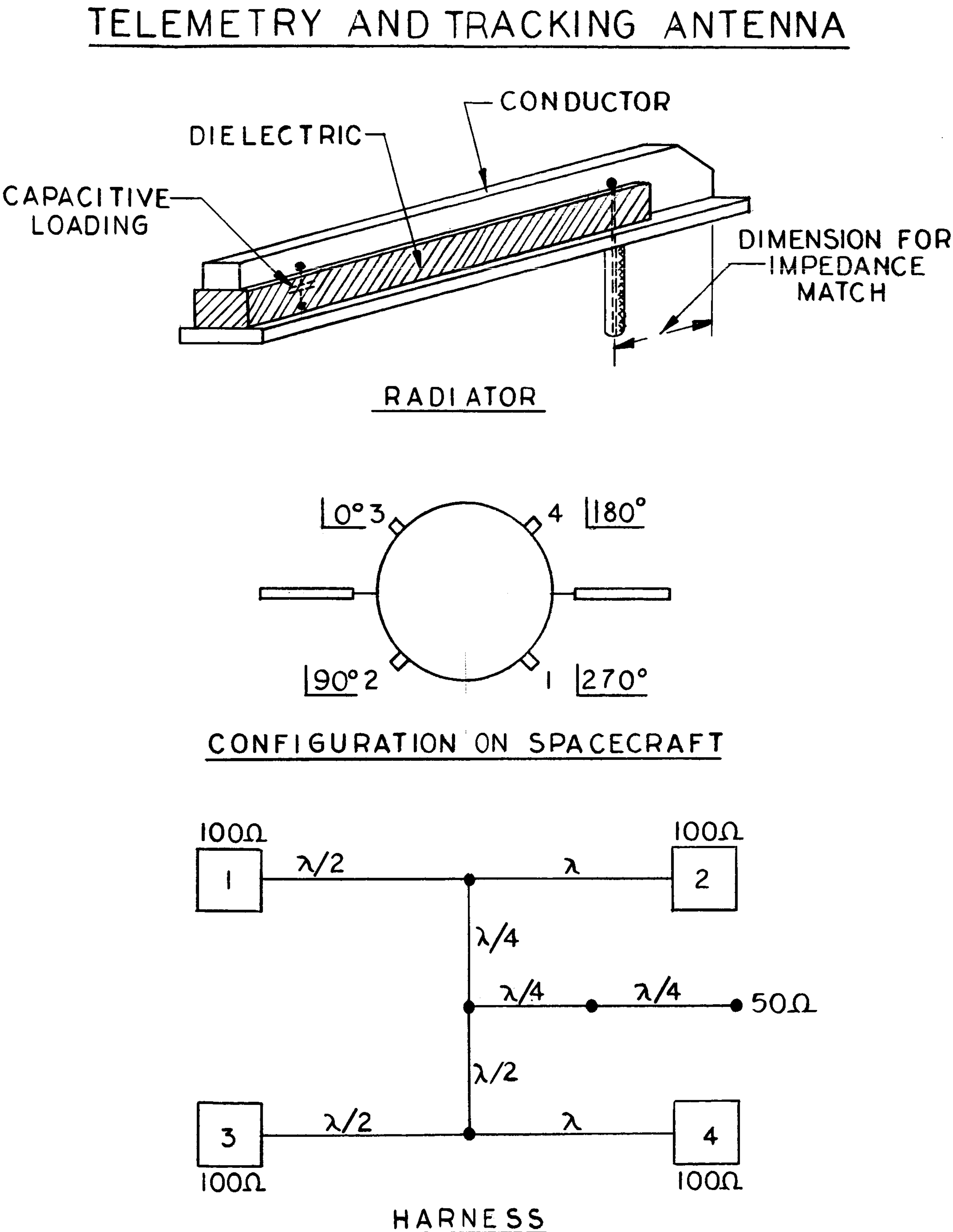


Figure 12. Telemetry and Tracking Antenna

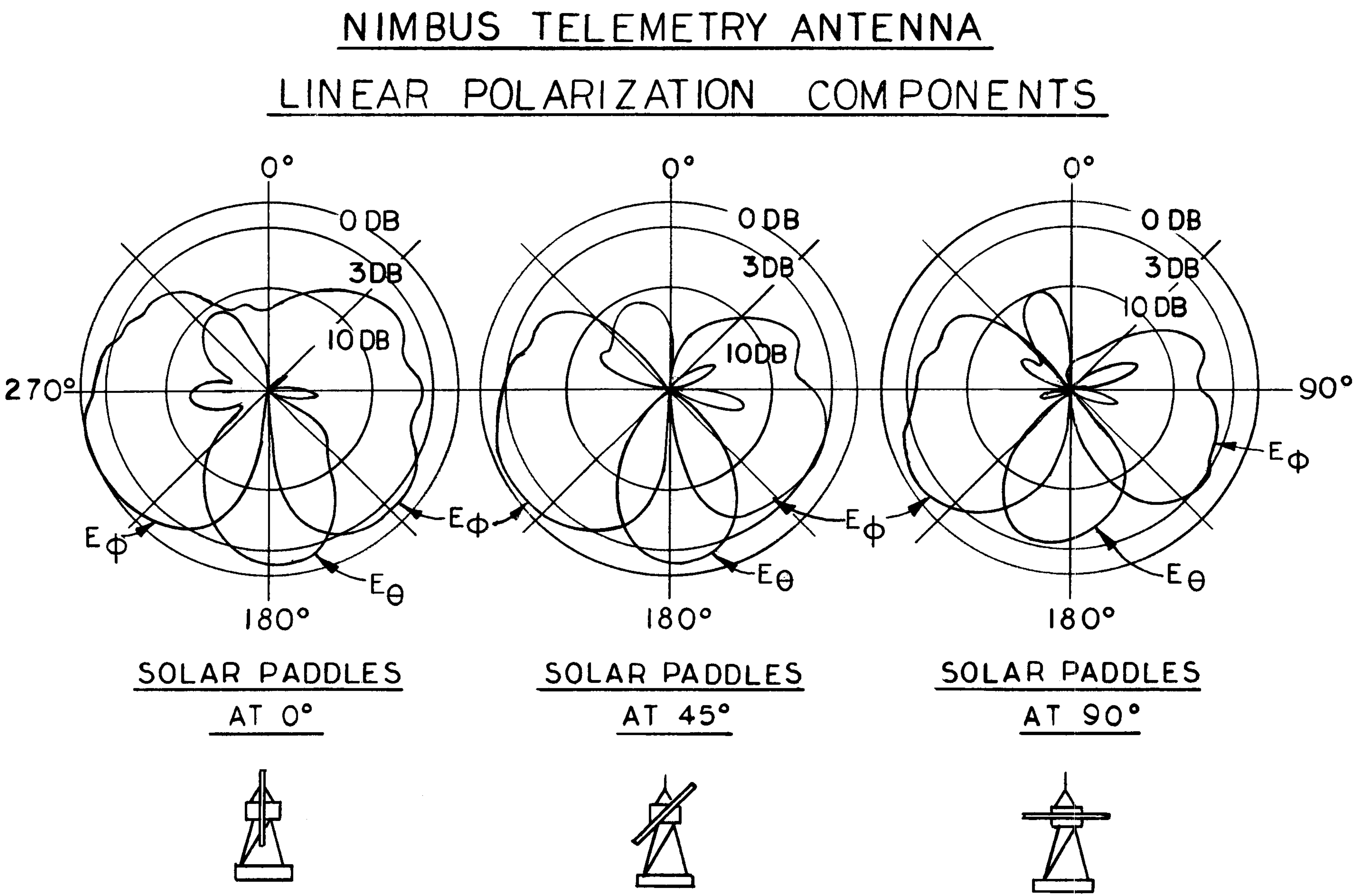


Figure 13. Nimbus Telemetry Antenna

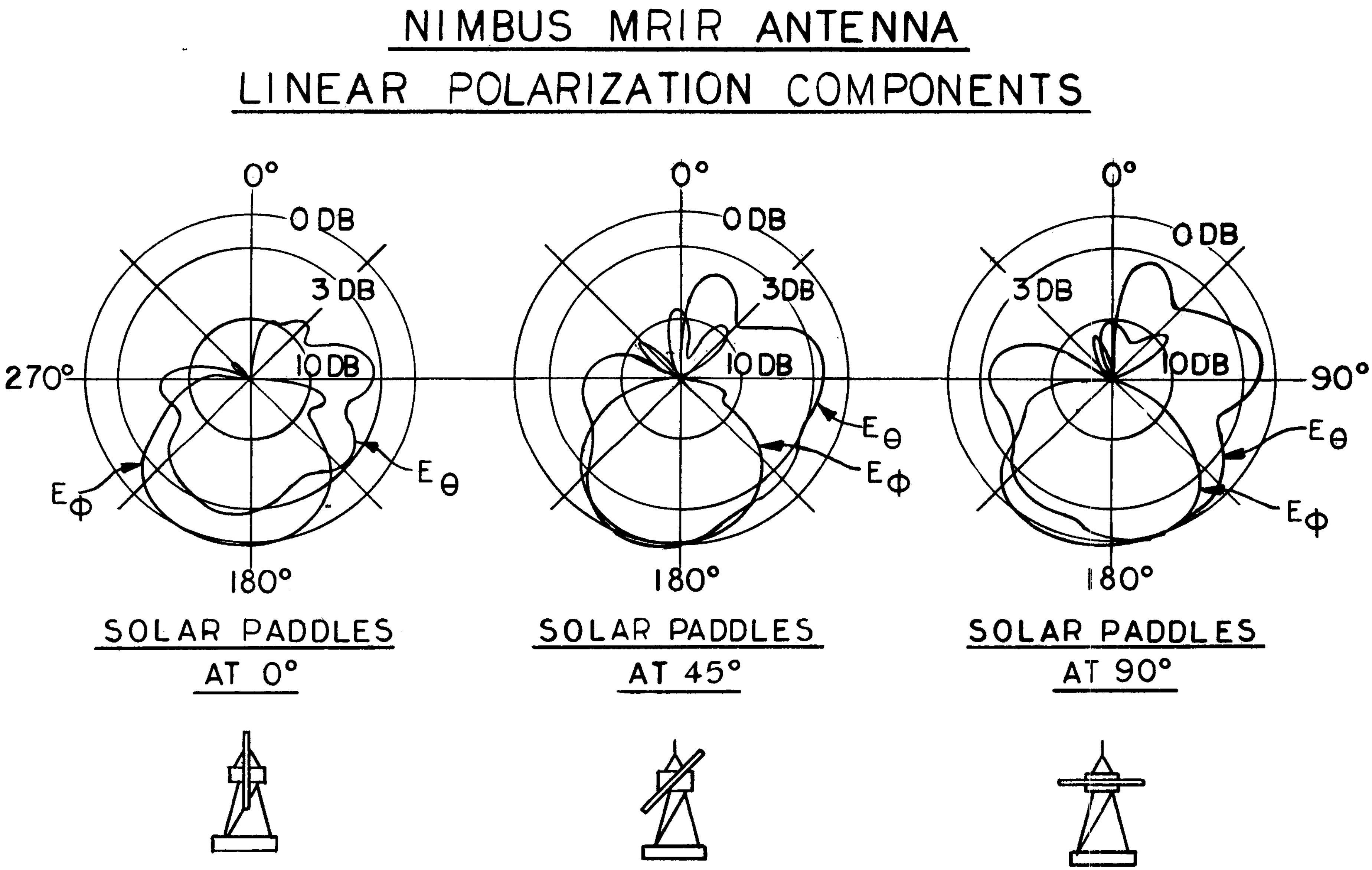
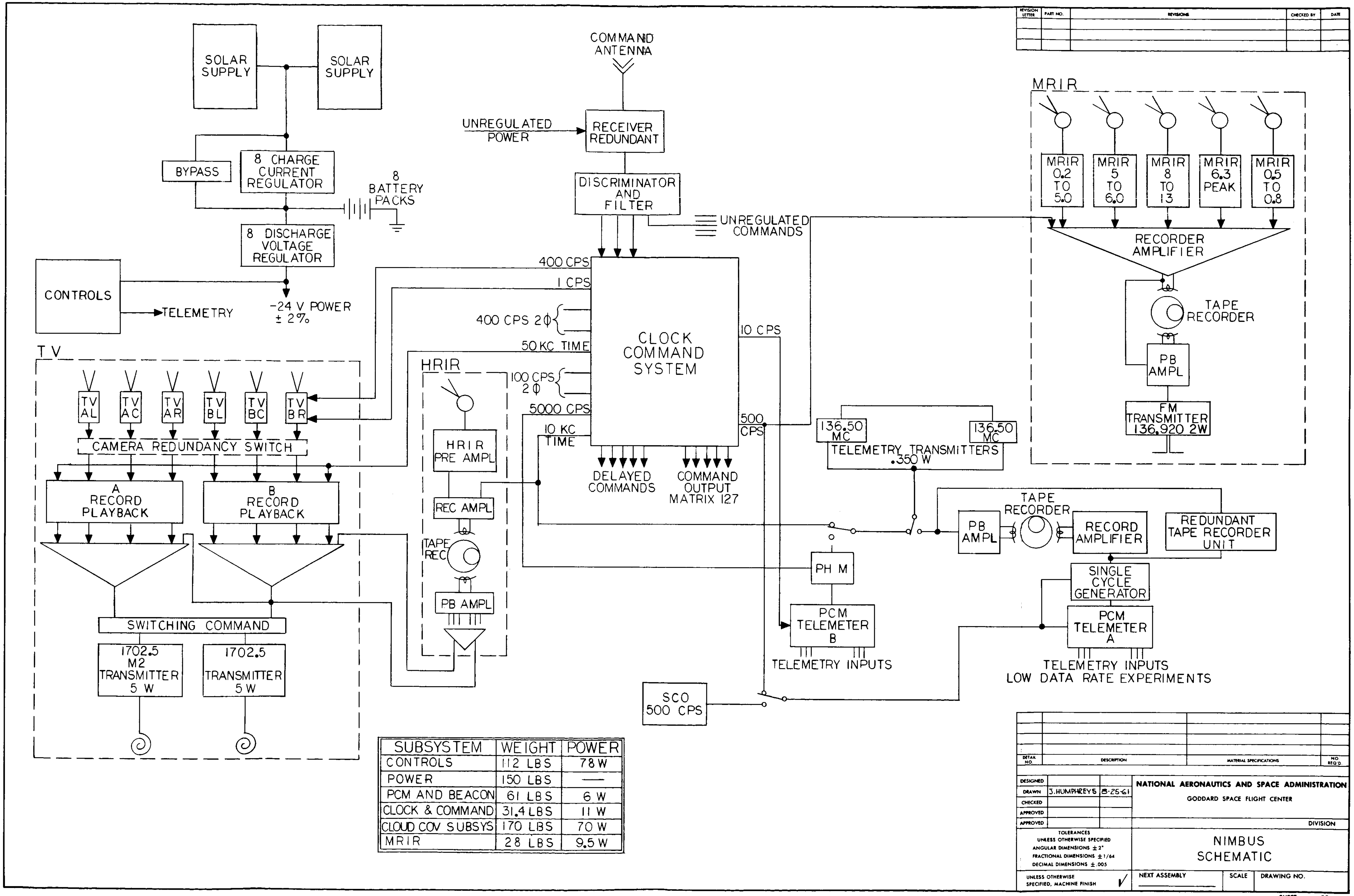


Figure 14. Nimbus MRIR Antenna



SUBSYSTEM	WEIGHT	POWER
CONTROLS	112 LBS	78 W
POWER	150 LBS	—
PCM AND BEACON	61 LBS	6 W
CLOCK & COMMAND	31.4 LBS	11 W
CLOUD COV SUBSYS	170 LBS	70 W
MRIR	28 LBS	9.5 W

Figure 15. Nimbus Schematic

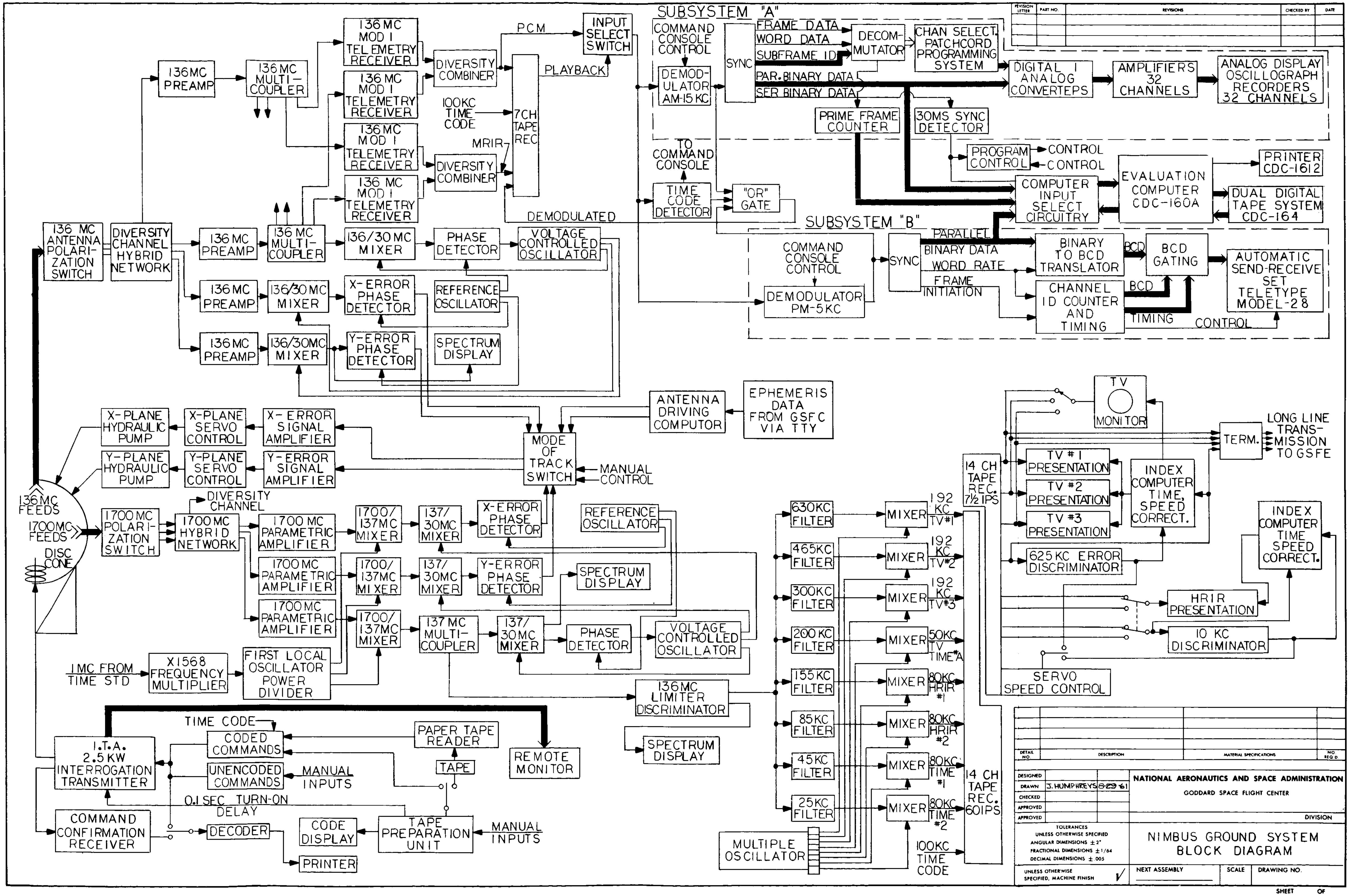


Figure 16. Nimbus Ground System

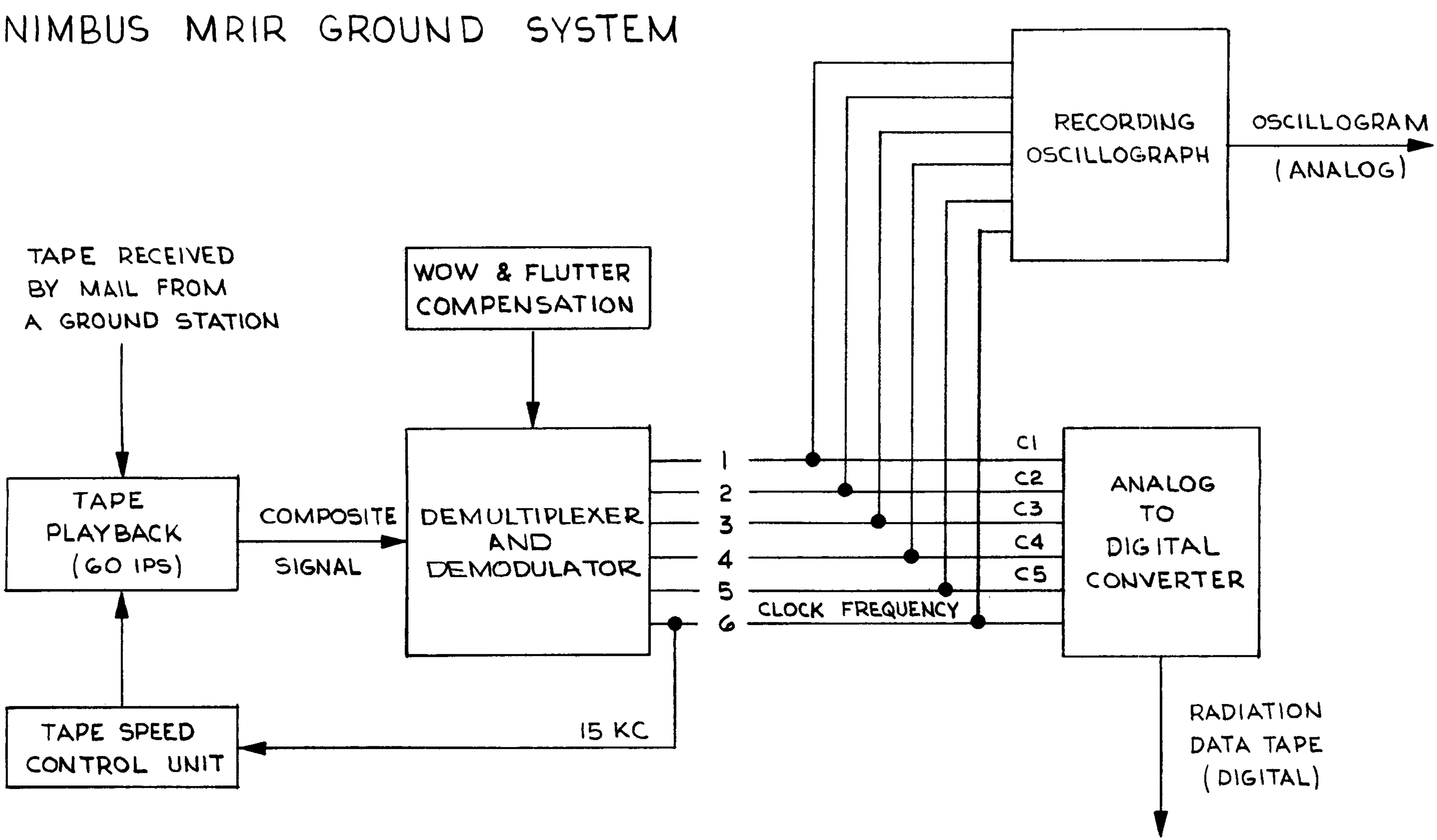


Figure 17. Nimbus MRIR Ground System